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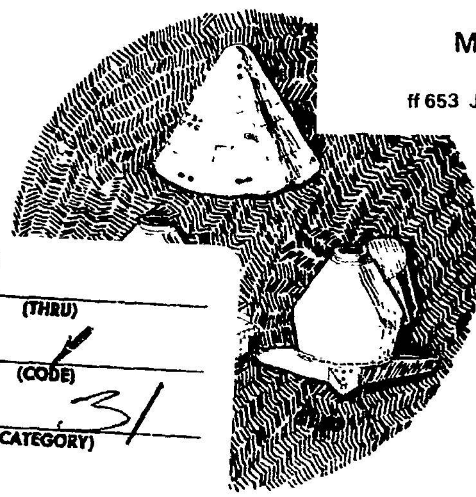

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STUDY OF A

Renovated Command Module Laboratory and Renovated Command Module

Contract No. NAS9-6445

FINAL REPORT

Volume III

Subsystems Analysis

NORTH AMERICAN AVIATION, INC. SPACE and INFORMATION SYSTEMS DIVISION

NASA CR 92157

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STUDY OF A RENOVATED COMMAND MODULE LABORATORY AND RENOVATED COMMAND MODULE

Final Report

VOLUME III - SUBSYSTEMS ANALYSIS

15 DECEMBER 1966

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NORTH AMERICAN AVIATION, INC.
SPACE and INFORMATION SYSTEMS DIVISION

FOREWORD

This document is submitted by the Space and Information Systems Division of North American Aviation, Inc., to the National Aeronautics and Space Administration Manned Spacecraft Center in parital fulfillment of the final reporting requirements of Contract NAS 9-6445, "Study of a Renovated Command Module Laboratory and Renovated Command Module."

The final report has been prepared in a series of five volumes as listed below.

Volume I	Summary	SID 66-1853-1
Volume II	Mission System Performance and Configuration Analysis	SID 66-1853-2
Volume III	Subsystems Analysis	SID 66-1853-3
Volume IV	Resources Requirements Analysis	SID 66-1853-4
Volume V	Cost Analysis (Limited Access)	SID 66-1853-5

S&ID acknowledges the voluntary technical contributions made to this study by a number of companies. The Avco Corporation contributed ablator data which were used as a basis for determining the feasibility of heat shield renovation. A report covering the data provided by Avco is included as an appendix to Volume III.

A. C. Electronics Division of General Motors Corporation supplied data on technical problems associated with renovating the Apollo G&N system and estimated costs.

The Defense Programs Division of General Electric Company provided characteristic data on G.E.'s active space pointing systems.

Westinghouse Electric Corporation provided data on rendezvous radar and transponder characteristics.

The Aeronautical Division of Honeywell, Inc., provided renovation data on the Apollo Block II stabilization and control system and associated costs.

The Autonetics Division of North American Aviation, Inc., provided data on an alternative guidance and navigation system and estimated costs.

Cost information and general renovation requirements on individual components were also provided by numerous other suppliers.

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ILLUSTRATIONS

TABLES

I. INTRODUCTION, APPROACH, AND SUMMARY

In addition to the obvious economic benefits of renovating recovered space program hardware for reuse, this class of hardware may offer a significant schedule advantage by merit of its availability. In some cases, this availability permits performance of missions that would not otherwise be possible. This document covers the results of the subsystem analysis conducted primarily to ascertain the technical feasibility of the renovated command module concept.

The feasibility of renovating a recovered Apollo command module for reuse is technically dependent on the postrecovery condition or status of the CM subsystems, materials, and structure. Also of significance is that renovation required to return the recovered equipment to condition acceptable for reuse in the specific missions considered. These two aspects formed the primary study areas in the subsystem analysis.

Two general renovated command module (RCM) applications considered in the study an (1) RCM Spacecraft (using only Block II subsystems) for use in an AAP operational CM in a fourteen-day, low-altitude, low-inclination earth orbit, and (2) RCM Laboratories (using Block I and II subsystems) based on the four reference missions described in Volume II. No RCM Laboratory functional capability was specified. Instead, a spectrum of system functions (from complete dependence on the supporting Apollo command-service module (CSM) to complete laboratory independence) was used to provide criteria for the development of subsystem building blocks "shopping list." The purpose of the "shopping list" is to provide a source of space-rated, available subsystems and components from which the subsystems complement of the RCM Laboratory could be generated when specific mission requirements were defined.

The approach taken in the subsystem analysis is shown in the logic diagram of Figure 1. Beginning with Apollo Spacecraft 009 and 011 and Gemini V, VIA, and VIII preflight and postflight subsystem data, the anticipated Apollo CM subsystem operational degradation was developed. In this process, the operational environment profile and postrecovery operations were reviewed and the resulting physical effects on the subsystems and their components were ascertained. These, in turn, were related to functional performance degradation. Nominal mission conditions were considered and a degree of technical judgment was applied in the performance of this status analysis.

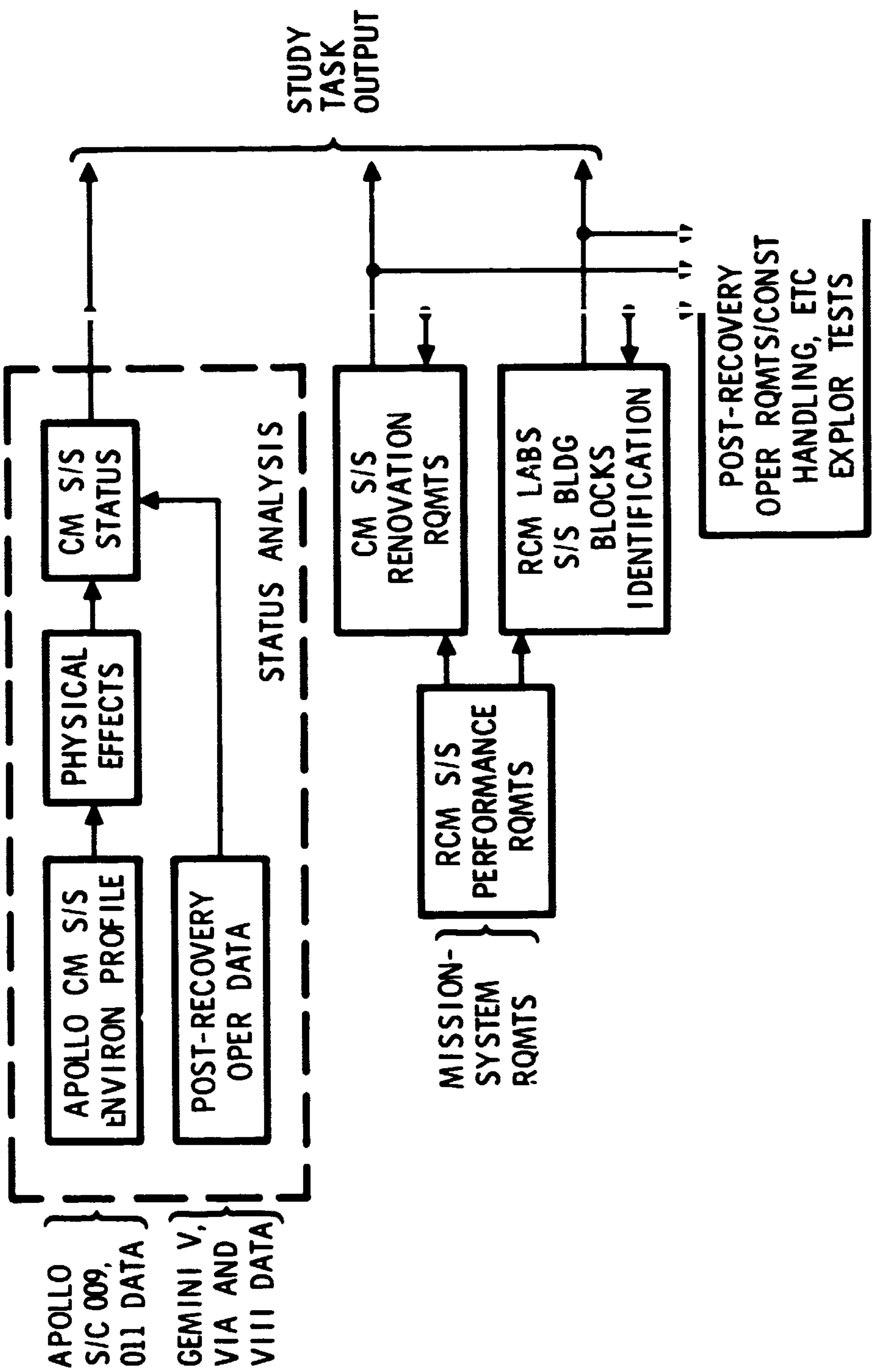


Figure 1. Subsystem Analysis Approach

On the basis of the required RCM subsystem performance and environmental requirements, the necessary renovation of the Apollo CM subsystems was detailed. The subsystem components were categorized as to "test and reuse," "refurbish," or "replace" renovation actions; a percentage of original cost was estimated for the required refurbishment.

RCM Laboratory subsystem building blocks were identified from those space-qualified items which might be incorporated with minimum testing, development, etc. Those requirements and constraints on the post-recovery operations which assist or make possible the subsequent renovations were also determined.

Apollo Spacecraft 011 data constituted the basepoint of the status analysis activity. Figures 2 and 3 show Spacecraft 011 after recovery and transportation to Downey. As can be seen, the crew compartment heat shield suffered only minor degradation in the form of discoloration and slight charring, with the exception of the RCS roll thrustor areas, in which serious charring was caused by the dump-burn of the RCS propellant prior to splashdown. This subject is further discussed in the following detailed sections.

Figure 2 shows damage to the drogue chute cases incurred during riser deployment. Relocation of the risers on Block II will eliminate this type of damage, permitting reuse of the cases.

The status analysis revealed that sea-water corrosion was a major potential cause of degradation. It also revealed that there was no danger of structural or material fatigue failure, safety-factor reduction, or temperature degradation. Landing impact causes fluid system seal degradation, producing internal and external leakage. The impact load also affects the mechanical adjustment of devices such as calibrated regulators, meters, tuned cavities, etc. The degradation of the crew compartment head shield is minor with the exception of a few small areas. The aft heat shield may suffer various degrees of degradation, dependent upon the mission and reentry conditions.

On the basis of the number of subsystem components involved, and the assemblies where large numbers of small and low-cost items are integrated into a subsystem part (such as control and instrument panels), it was determined that renovation for the RCM spacecraft requires approximately 25 percent of the spacecraft components to be replaced and 47 percent refurbished, while 28 percent can be tested and reused.

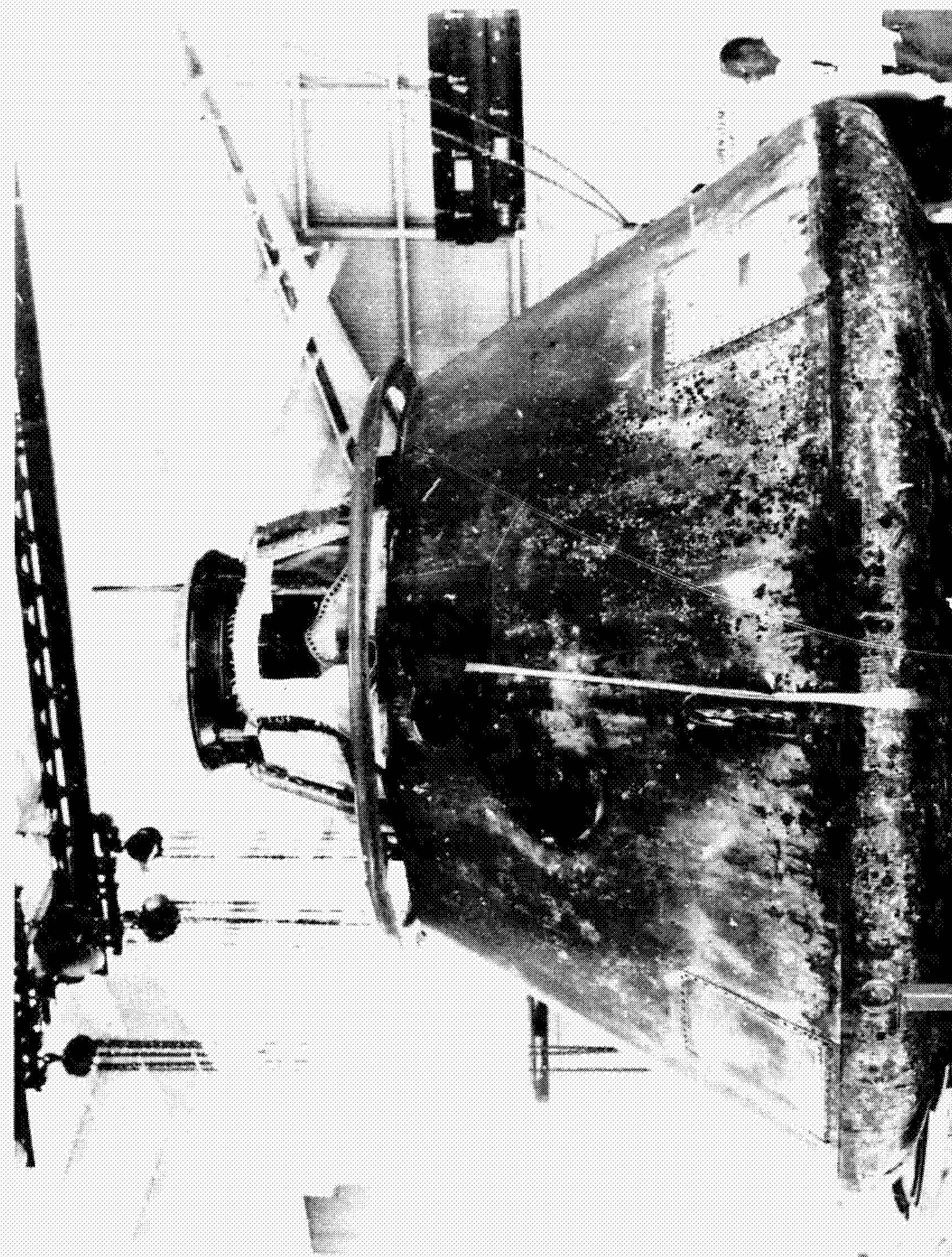


Figure 2. Apollo Spacecraft 011 After Recovery and Transportation to Downey, Region of Reentry Stagnation

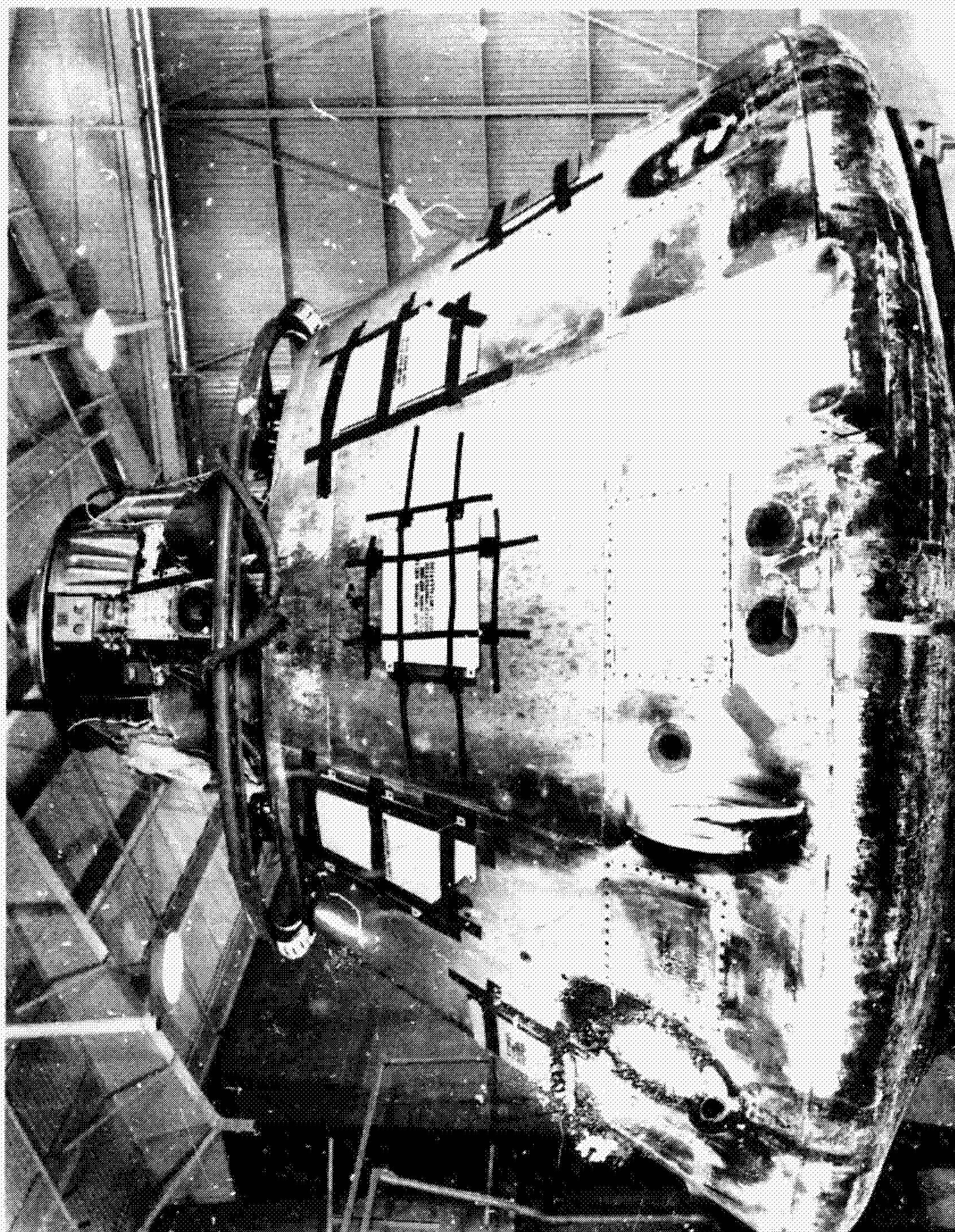


Figure 3. Apollo Spacecraft 011 After Recovery and Transportation to Downey, Leeward Region

II. STATUS ANALYSIS SUMMARY

As stated in Section I, the major item in determining the technical feasibility of Apollo CM renovation is the status or condition of the CM subsystems upon recovery after an operational mission. Accordingly, this matter was given primary attention in the subsystem analysis and was identified as the status analysis. Latter portions of this section describe the CM environmental profile data and the Apollo/Gemini postrecovery data which constituted the basis for the detailed status analysis conducted in each subsystem area. The results of these analyses are covered in the following sections of this volume. Considering the potentially susceptible materials used in each subsystem, the physical effects of the operational environment resulting from a nominal mission were determined. These were then translated into component and subsystem performance capability changes, and in combined consideration with the effects of the postrecovery operations, the anticipated CM subsystem status was derived. In the majority of cases, no rigorous path leading to these desired conclusions was available. The conservative and considered application of technical judgment was required, and used, in the conduct of the status analysis as well as a subsequent phase of the subsystem analysis: CM subsystem renovation requirements determination.

Results of the status analysis indicate that salt water corrosion presents the most serious degradation factor. Exposure occurs at splashdown, when nominally the sea water penetrates the interspace between the heat shields and the pressure vessel, and enters the toroidal equipment bay. No design provisions have been taken to seal against this invasion, and as a result sea water also penetrates the core field of the stainless steel honeycomb substrata of the heat shields. As can be seen From Table 1 the honeycomb materials (PH 14-8 MO and 17-4 PH) are highly susceptible to corrosion. Laboratory tests conducted during the study revealed that flushing and neutralizing of this structure prior to 24 hours after splashdown was required if the structure was to be considered for reuse. Beyond this time span, the corrosive action progresses to the point where pitting has occurred through the coating of brazing material to the core material. Under these conditions, flushing and neutralization is considered ineffective, since the galvanic action of this couple of dissimilar metals continues and eventually causes failure of the structure. These laboratory tests and purging tests, resulted in a recommended purging procedure discussed in detail in Section III.

Table 1. Corrosion Due to Sea Water Contact

Material	CM Application
HIGHLY RESISTANT	
Titanium 6AL-4V Inconel 718 Steel 304L	RCS tanks Oxygen tanks Tubing
RESISTANT	
Aluminum 6061 Aluminum 5052	ECS tanks, tubing HCB core
HIGHLY SUSCEPTIBLE	
Steel PH 14-8 MO Steel 17-4 PH Aluminum 2014 T6	Brazed HCB* Edge members* Bonded HCBΔ
ΔPressure vessel *Heat shield	

A simplified cumulative fatigue analysis was performed using conservative assumptions to justify the simplifications made. Extremely low structural damping was assumed. Each of the loads was applied twenty times at maximum level to simulate this mildly damped structure. The most fatigue-sensitive material used was considered throughout the analysis (epoxy-phenolic adhesive), and all loads were applied at the maximum design level. The result demonstrated that no fatigue damage or safety-factor reduction will occur, even after the performance of the subsequent RCM mission.

Table 2 shows the design temperatures for the various CM materials. Also shown is the threshold temperature for the materials beyond which changes to the material physical properties occur. Coupled with the fact that all Spacecraft (S/C) 011 temperatures were below the design levels, it can be seen that a nominal mission would result in no temperature degradation to the CM materials.

Fluid system seals may be degraded by the landing impact, thus resulting in internal and external leakage. It was concluded, however, that elastomers employed in a manner where their elastic properties constitute

an essential characteristic of their function will be replaced during subsystem renovation. Therefore, this degradation in itself is academic. The obvious secondary effects of external leakage (corrosion and contamination) can be offset easily during postrecovery operations.

Table 2. Temperature Effects on CM Material

Material	CM Application	Design Temperature (°F)	Temperature Threshold for Mechanical Property Changes (°F)
Steel: PH 14-8 MO 17-4PH 304L	Brazed HCB Edge member Tubing	600 600 600	1000 1100 1200
Aluminum: 2014T6 5052H39 6061T6	Bonded HCB HCB core ECS tanks, tubing	200 200 200	250 300 300
Titanium 6Al-4V	RCS tanks	125	950
Inconel 718	Oxygen tanks	200	1200
Epoxy-phenolic Adhesive	Heat shield Bond	600	700
	Inner structure Adhesive	250	700

Degradation to the heat shield is dependent on the reentry conditions. It was determined that the virgin ablative material remaining after a nominal earth orbital mission is adequate for a subsequent RCM spacecraft mission; however, the material remaining after a nominal lunar mission is not sufficient for reuse. Superimposed upon these considerations are those that involve questions of the thermostructural and aerothermodynamic qualities of the ablative material after mission and reentry environment exposure. Conservative conclusions indicate that the aft heat shield can be renovated by the complete replacement of the used ablative material with new material.

This approach has been already demonstrated (Class 3 Apollo heat shiel' repair) as feasible and, although cost savings are not as great as woul be achieved in some approach involving less renovations a significant cost saving results in the reuse of the stainless steel honeycomb substrate. More important, the renovation lead time is less than six months, whereas approximately two years lead time is required for a new heat shield.

The status analysis also revealed that no significant degradation to the CM subsystems is anticipated due to exposure to the radiation, vacuum, or meteoroid environment.

APOLLO AND GEMINI POSTRECOVERY DATA

Potentially applicable Apollo and Gemini documentation was compiled and reviewed to determine the possibility of ascertaining the postrecovery status of the command module subsystems by comparing the applicable preflight test results data with the postrecovery test/inspection results data, and determine the effects of postrecovery operations on the CM subsystems and components.

Current postrecovery test and inspection procedures were found to be inadequate to support the first objective. This inadequacy did not, however, impose any serious obstacle to the RCM program since other practical limitations were found to make either evaluation and/or renovation of any assembled subsystem neither technically nor economically feasible. In the end, these evaluations and/or renovation must be performed at the major component level.

The postrecovery operations effects were found to be degradation and damage caused primarily by corrosion, contamination, and handling. Existing recovery procedures have been designed for safe, efficient recovery of the spacecraft, but these procedures contain no general provisions to minimize these degradation effects since renovation was not considered in their formulation. Early in the RCM study, it was deemed necessary to develop preliminary recommendations for possible procedural changes to reduce these degrading effects. The resulting recommended recovery events sequence as presented herein was subsequently reviewed by the respective subsystem analysts and the various support activities. The resulting recommended recovery operations procedure is presented in Volume IV of this report.

The following documents were compiled and reviewed to support the subsystem status analysis.

1. MSC-G-R-65-4. "Gemini Program Mission Report - Gemini V" (October 1965)

2. MSC-G-R-66-2. "Gemini Program Mission Report - Gemini VIA" (January 1966)

3. MSC-G-R-66-4. "Gemini Program Mission Report - Gemini VIII" (April 1966)

4. MAC Report PS-186. "Corrosion Control Procedures for Recovered Spacecraft" (Revised 11 October 1965)

5. Project Apollo Flight-Test Report - Spacecraft 009, NAA S&ID SID 66-150 (August 1966)

6. Post-Flight Evaluation, Environmental Control System - Apollo S/C 009, AiResearch Report, SS-1917-R (13 July 1966)

7. Apollo Recovery Operations Handbook, S/C 011. NAA S&ID SID 66-326 (15 June 1966).

8. ATR 521060. "Block I, Post Recovery Test, Baseline, at Downey, Command Module, S/C 011 and 012" (August 1966)

These documents were considered to be representative of the types of applicable data currently being generated during the Gemini and Apollo Programs. The listed documents consist of program reports or procedures relating to Gemini Missions V, VIA, and VIII and Saturn-Apollo Mission SA 201 (S/C 009) and published by NASA/MSC, McDonnell, NAA/S&ID and AiResearch. These reviews and related subsystem status analyses were further augmented by additional unpublished reports and letters, such as the certification test data for the S/C 009 and 011 flight readiness reviews (FRR's), S/C 009 post-recovery testing ATR's and TPS's, and several NASA/MSC and McDonnell documents relating to either corrosion control or to Gemini and/or Apollo equipment reuse or refurbishment.

Postrecovery Inspection/Test Results Data

Review of the support documentation showed that the current Gemini and/or Apollo postrecovery inspections/test results data were inadequate for comparison with applicable preflight test results data to determine subsystem status. The present postrecovery test procedures are primarily designed to (1) evaluate anomalies or failures discovered in flight, (2) determine the occurrence of visually-detectable physical damage, and (3) uncover additional failures or anomalies not previously detected. The resulting types of postrecovery test data, therefore, provided limited useful support in determining subsystem status, particularly with respect to marginal or partial performance degradations within specific equipments.

It is possible to establish postrecovery test procedures that provide for measurement of all critical performance parameters. These measurements can then be compared with the like preflight acceptance data to detect most significant performance degradations at the component level. It is doubtful, however, that detection of marginal degradations would enable judgements to be made of the future performance (and reliability) of these components. Such capability is particularly lacking on components that are subject to wearout.

The current component and/or subsystem tests for the Apollo hardware were devised to demonstrate only the specified operational duration of the applicable hardware. This testing does not provide data required for definition of the wearout failure distributions for the equipment. This data can only be acquired by testing several of each of these equipments to failure. Such a test program is prohibitively expensive, and it is doubtful if the possible cost savings that may result could offset increased cost. The potential for additonal life testing is further complicated by the questionable capability of duplicating the combined effects of environments and operations for the projected two-mission utilization.

A possible alternative to the additional testing discussed is analytical determination or postulation of the characteristic wearout of the applicable equipment which would require an extensive study effort by the contractor and the respective subcontractors and supplier. This study could utilize empirical data on similar components and/or materials that may have been exposed to either like or relatable operating conditions. Such an analytical activity would, however, most certainly be pushing the state of the art in many or all of the areas concerned. The results would, therefore, prove the acceptability for reuse of only those items that might contain unexpected and unduly conservative derating factors and design margins. It should be noted that this approach has been recommended herein for "requalification by similarity" for limited, specific items that were not considered to contain significant or critical wearout failure modes.

The preceding discussion covered the problems inherent in evaluating simple components (i.e., valves, pumps, relays, etc.) for reuse, ostensibly, without rework. The desired possibility of postrecovery test procedures to determine the status and reuse capabilities of entire subsystems is more difficult. The Apollo command module subsystems are characteristically complex, using and producing multiple and varied inputs and outputs. They also utilize numerous redundant and alternate modes to perform their functions. Equipment of either equal or greater complexity than the Apollo automatic checkout equipment (ACE) currently being used would be required to measure every input and output parameter. Even the ACE cannot determine the existence of certain failures such as an individual check

valve failure in a parallel, redundant application. The nearly infinite number of interactions that exist within each of these subsystems further complicates this evaluation procedure. Finally, for this test procedure to be of practical value, it would be necessary to subsequently retain and maintain an entire subsystem intact and assembled within the command module for the entire renovation period. This would require that the CM be stored and reworked in a clean, environmentally controlled area in order to minimize potential system degradations.

In view of the problems and costs associated with implementation of either of the two alternative postrecovery test procedures previously discussed, neither method was deemed to be practically feasible. It was, therefore, concluded that a more practical approach would have to be used. This approach consisted of revising the postrecovery procedures to reduce potential corrosion, contamination, damage, and subsequent removal of the equipment to permit, as applicable (1) testing and reuse, (2) refurbishment, or (3) replacement. Decisions concerning specific equipment are discussed and presented in later sections of this report.

Gemini and Apollo Flight Anomalies

A compilation was made of the inflight and postflight anomalies reported in the mission reports for the Gemini V, VIA, and VIII, and the Apollo S/C 009 and 011 flights. The results of this compilation were summarized and are presented in Table 3. The purpose of this activity was to attempt to determine any significant trends in these anomalies that might be of potential use during RCM study. With exception of repeated instances of corroded coaxial switches and connectors, open fusistor/fuses in the pyrotechnic circuits, and residue coating on windows, the reported anomalies appear to be either random occurrences or the result of design deficiencies. Several anomalies were reported on tape records the Gemini flights; however a different type recorder is used on Apollo. Only one anomaly was reported which could have potentially affected crew safety. The electrical malfunction which resulted in a flight control anomaly on Gemini VIII caused a mission abort. In general, no significant anomaly trends were found that might have been useful in establishing the refurbishment requirements.

A further attempt was made to determine if any relationship existed between the total mission hours, number of inflight anomalies, and number of postflight anomalies. The resulting values were:

1. Gemini V: 191 hours - 12 hours inflight, 9 hours postflight,

2. Gemini VIA: 25.9 hours - 4 hours inflight, 9 hours postflight

3. Gemini VIII: 10.7 hours (aborted) -4 hours inflight - 15 hours postflight.

Table 3. Anomalies (Inflight/Postflight)

Subsystem \ Spacecraft	Gemini V	Gemini VIA	Gemini VIII	Apollo S/C 009	Apollo S/C 011	Primary Degradation Factor Comparisons
Communications	1/0	1/1	0/2			Gemini: Corroded coaxial switches and connectors Apollo: Antenna-sensitive to sea water
Instrumentation and recording	3/0	2/0	2/0	7/0		Gemini: Tape recorder - mechanical deficiency Apollo: Diode failure, defective bonding, plus various unknowns
Guidance and control	5/0	0/1	0/4			Gemini: Electrical malfunction (caused Gemini VIII abort) Apollo: Power failure
Electrical	0/1	1/2	1/3	3/0		Gemini: Corroded connectors Apollo: Short circuits during entry operations
Propulsion	2/0			2/0		Gemini: OMS heater malfunction Apollo: (1) Helium ingestion (2) Unknown
Environmental control		0/1	0/1	1/1		Gemini: O_2 leakage, loose rubber plug Apollo: Design deficiency, impact damage
Reaction control				2/4		Gemini: None Apollo: Oxidizer isolation valve failed
Structure	0/4	1/1	0/4	0/2		Gemini: Windows coated with residue Apollo: Windows coated with residue
Total	11/5	5/6	3/14	15/7		

These data fail to indicate any significant trends that might be potentially useful to the RCM study. They appear to indicate, however, a lack of complete knowledge of the inflight anomalies to support the postflight evaluation (note Gemini V results) and/or a lack of consistent reporting and inspection procedures. The larger number of items reported on in the Gemini VIII postflight evaluation might also be indicative of either greater damage due to loads imposed during the mission (the vehicle achieved a roll rate of approximately 300 degrees per second) or a more thorough postflight examination.

Postrecovery Operations Effects

Review of the postrecovery procedures and results data previously discussed leads to the following conclusion and recommendation. Existing Apollo postflight recovery and test procedures should be revised to:

1. Minimize potential damage, contamination and corrosion to potentially reusable hardware keeping in mind that the primary purpose of the current procedures is to recover data required to qualify the Apollo for an LOR mission

2. Provide reasonable assurance that applicable critical components have the capability to perform satisfactorily when exposed to at least the nominal environments of the subsequent, planned mission

3. Ensure positive means of reidentification of equipment to eliminate their possible reuse on more critical missions

4. Eliminate separations of all possible interfaces by development of test procedures specifically designed to determine the status of subsystems, or significant portions thereof.

Item 4 is particularly desirable and advantageous where welded, soldered, and bonded joints are used, but may not be possible where analysis indicates a need for replacement or refurbishment due to potential wearout, replacement of elastomers, or inability to check out specific critical components.

To assist in satisfying the preceding conditions, listings were prepared of the reported Gemini recovery events (Table 4). The preceding data was reviewed by each of the effected activities and the recommended recovery events sequence for the Apollo RCM was developed (Table 5).

Table 4. Gemini Recovery Events Sequence

Event	Ground Elapsed Time
GEMINI V	
1. Touchdown (29 August 1965)	190:55:14
2. Swimmer team deployed	191:38
3. Flotation collar attached and inflated	191:45
4. Swimmer - flight crew voice contact made	
5. Crew egressed (left hatch) and boarded helicopter	191:58
6. Carrier retrieved spacecraft	194:50
7. Specified postretrieval procedures performed:	

Equipments removed, cleaned and packaged aboard prime recovery vehicle included: Voice tape recorder, inertial measurement unit (IMU), attitude control maneuver electronics (ACME), computer, auxiliary computer power unit (ACPU), horizon sensor electronics, PCM tape recorder, programmer, instrumentation package No. 2, high-level and low-level multiplexers and undefined flight crew and experiment equipments.

All on-board film and certain equipments were expedited to Cape Kennedy and Houston by special flights from carrier.

Visual Inspections and observations were:

a. No excessive heating effects

b. Two deep gouges in heat shield

c. Both windows had moisture between glass layers. Protective covers were emplaced

Table 4. Gemini Recovery Events Sequence (Cont)

Event	Ground Elapsed Time
d. Hoist loop door and recovery light funtioned properly.	
e. Left-hand hatch was closed but not locked. Right-hand hatch was locked. Both hatch seals appeared in excellent condition	
f. Spacecraft interior was exceptionally clean and all equipment was stowed	
g. All spacecraft power, except recovery light, was off	
h. Drogue mortor was safed (pins) and ejection seat D-ring was stowed	
i. Moisture found in both footwells	
8. Spacecraft off-loaded at Mayport Naval Station (30 August 1965)	
9. RCS deactivated (took approximately 9 hours)	
GEMINI VI	
1. Touchdown (16 December 1965)	25:51:24
2. Swimmer team deployed	26:14:34
3. Flotation collar attached and inflated	26:21:34
4. Flight crew stayed in S/C - opened left hatch	26:33:34
5. Spacecraft on carrier and secured	26:57:34
6. Postretrieval procedures (similar to GT-V) performed	- - -
7. Spacecraft off-loaded at Mayport (20 December 1965)	
8. RCS deactivation (took approximately 16 hours)	

Table 4. Gemini Recovery Events Sequence (Cont)

Event	Ground Elapsed Time
GEMINI VIII*	
1. Touchdown (17 March 1966)	10:41
2. Swimmer team deployed	10:54
3. Flotation collar attached and inflated	11:30
4. S/C hatches opened (flight crew returned to ship)	11:48
5. S/C secured on destroyer (U.S.S. Mason) (On-board films and tapes removed)	14:15
6. S/C off-loaded at Okinawa (18 March 1960)	31:29
7. RCS deactivated** (20 March 1960) - no time recorded	- - -
*Secondary impact area **For safety - C-130 transport to Mainland (pyrotechnic safing not noted, but assumed)	

Table 5. Preliminary Apollo RCM Recovery Events Sequence

Flight Crew to Perform Following Events Prior to Touchdown:

1. Obtain cabin air sample subsequent to reentry and prior to opening of steam vent line to atmosphere.
2. Subsequent to CM RCS purge system activation (helium pressure $\leq$ 100 psi) and prior to impact:
   a. Turn off CM RCS propellant isolation valve - A and B control switches
   b. Turn off either CM RCS dump or Logic switch
3. Shut down nonessential CM systems and, if time permits, record status.

Table 5. Preliminary Apollo RCM Recovery
Events Sequence (Cont)

Events Sequence at Recovery Area:

1. Touchdown
2. Deploy swimmer team
3. Swimmers plug heat shield vent hole, inspect CM, and attach and inflate flotation collar
4. Establish swimmer - flight crew voice contact
5. Flight crew deactivate recovery aids and record status
6. Flight crew recovery alternatives:
    a. Preferred: Flight crew remain in CM with hatches closed and postlanding ventilation system operating.
    b. Alternate: Flight crew egress if required by rough seas, astronaut illness, postlanding ventilation problem or spacecraft recovery delay. Use following sequence:
        (1) Open outer and inner crew compartment access hatches (or forward tunnel access hatch, if required). Swimmers maintain CM with hatch downwind, if possible.
        (2) Flight crew deactivate postlanding ventilation system.
        (3) Flight crew egress to life raft for helicopter retrieval.
        (4) Swimmers close and latch open hatches.
7. Swimmers disassemble HF Antenna and stow VHF Antenna (special tools required) and remain with CM until recovery ship arrives.
8. Hoist CM aboard recovery ship and monitor for radiation.
9. Place CM on recovery cradle.

Table 5. Preliminary Apollo RCM Recovery Events Sequence (Cont)

10. Wash down optics and CM RCS areas (if required).
11. Inspect, note, and record status of top deck pyrotechnic devices (safety and/or remove if required).
12. Obtain cabin air sample.
13. Open crew compartment outer and inner hatches (flight crew egress if still in CM).
14. Inspect for leakage of RCS propellants and instigate continuous monitoring.
15. Implace plastic shelter tent over CM and provide fresh, desiccated air under pressure.
16. Check control and display settings against checklist.
17. Shut down remaining CM systems.
18. Remove cameras, flight recorder and/or tapes and package for separate shipment.
19. Place desiccant increw compartment and close and secure inner hatch.
20. Remove aft heat shield and aft bulkhead thermal pads. Flush aft heat shield and toroidal equipment bay with boiler water. Insert purge fittings in aft heat shield honeycomb substrate. Vacuum purge aft heat shield with deionized water.
21. Vacuum purge aft heat shield with alcohol, then liquid freon.
22. Dry aft compartment equipment bay and heat shield with warm, dry air, insert desiccant, replace aft heat shield, and seal all heat shield openings.
23. Place protective covers over all CM windows and optics.
24. Return CM (sheltered) to forward area.

Table 5. Preliminary Apollo RCM Recovery Events Sequence (Cont)

Events Sequence at Forward Area:

1. Remove shelter and place CM on dock.
2. Remove CM to remote, safe area.
3. Safety pyrotechnic devices.
4. Decontaminate reaction control system.
5. Purge and dry environmental control system.
6. Reinstall access doors, hatches, seals and protective cover(s).
7. Transport CM to S&ID, Downey (air transport recommended)

Events Sequence at S&ID, Downey:

1. Place CM in sheltered, remote, safe area.
2. Remove top deck pyrotechnic devices and plug or cap resulting openings.
3. Remove heat shield access doors.
4. Remove RCS pyrotechnic devices. *
5. Remove outer crew compartment hatch and install protective cover on window of inner hatch.
6. Remove heat shield windows.
7. Remove aft heat shield.
8. Remove RCS engines, plugging all resulting openings.
9. Remove crew compartment heat shield.

Table 5. Preliminary Apollo RCM Recovery Events Sequence (Cont)

10. Clean exterior of CM.
11. Transfer CM to environmentally-controlled clean room for subsequent removal of equipment. **
*RCS and pyrotechnic safety aspects must be investigated. **Sequence of removals, criteria for checkout, and protection of interfaces and packaging and/or shipping requirements are not yet defined.

APOLLO COMMAND MODULE ENVIRONMENTAL PROFILE

One major cause of subsystem performance capability degradation is the environmental stresses imposed during mission operations. Accordingly, all available environmental data pertaining to the CM were reviewed. The applicable portions were then compiled and used as a base for the status analysis.

Since the study required that only nominal Apollo operational conditions be considered in the status analysis, the level value contained in the Apollo Environmental Specification (Reference 1) were adjusted to reflect this guideline and represent a realistic profile. The following portions of this section present this derived environmental profile and pertinent environmental observations from Apollo S/C 009 and 011 flights.

Environmental Profile Summary

The environmental conditions generally applicable to all the CM subsystems follow. Subsequent to these, discussions regarding the individual subsystems are presented.

Axial Acceleration

The axial acceleration depends on the launch vehicle. The Saturn IB (launch of S/C 009 and 011) produces a maximum sea level thrust of 1.6×10^6 pounds for the first stage, followed by 2×10^5 pounds of thrust from the single J-2 engine of the S-IVB. These combinations produced maximum g loads of 4.0 and 2.7 g, respectively, as compared to the 4.9 g load expected for the Saturn V configuration just before S-II ignition. The time profile obtained from Reference 4 is presented as Figure 4.

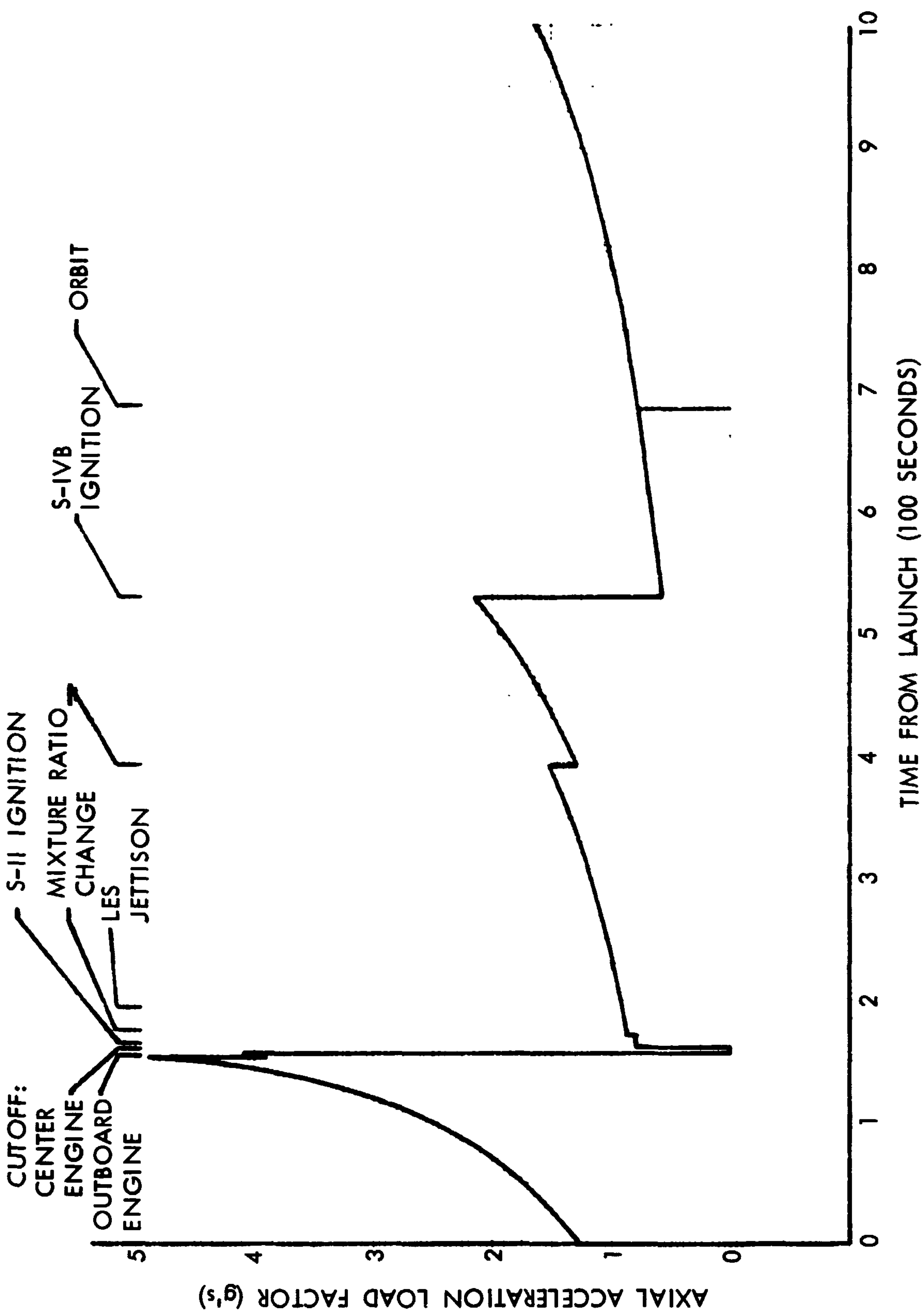


Figure 4. [illegible]urn V Launch Axial Acceleration

Acceleration

Figure 5 presents the maximum spectral densities of the accelerations expected on the structural subsystems. These curves were obtained from Reference 1.

Vibration

Vibration upon launch is expected to be the maximum condition. These levels of vibration are related to the sound pressure levels shown in Figure 6 and obtained from Reference 1.

Temperature and Static Pressure

The temperature of the space and ground environment are external parameters as presented in Table 6. Detailed air temperature profiles are as presented in the U.S. Standard 1962 Atmosphere applicable to altitudes up to 300,000 feet where direct conductive heat transfer to a vehicle is significant (except on entry at high velocity).

Table 6. Ground and Surface Air Temperature Extremes

Altitude	Storage (min)	KSC (min)	KSC (max)	Criteria
Ground	-10 F	35 F	170 F	Not exceeded by more than 10% of most extreme month (3 days) at location
3 feet	-15 F	28 F	110 F	

Expected and acceptable temperature range profiles for the command module crew compartment are presented in Figure 7.

Heat Shield

The heat shield in early Apollo flights is heavily instrumented for temperature, pressure, heat flux, char depth and ablation. Figure 8 presents a list of measurement points on the S/C 009 heat shield, with theoretical (specification) values for the peak levels expected.

Temperature-time histories were computed for entry according to the S/C 009 trajectory and are presented in Reference 4. Higher heating is expected for S/C 011 and for lunar entry, however, the S/C 009 and S/C 011 heating should bracket the range of conditions for orbital entry.

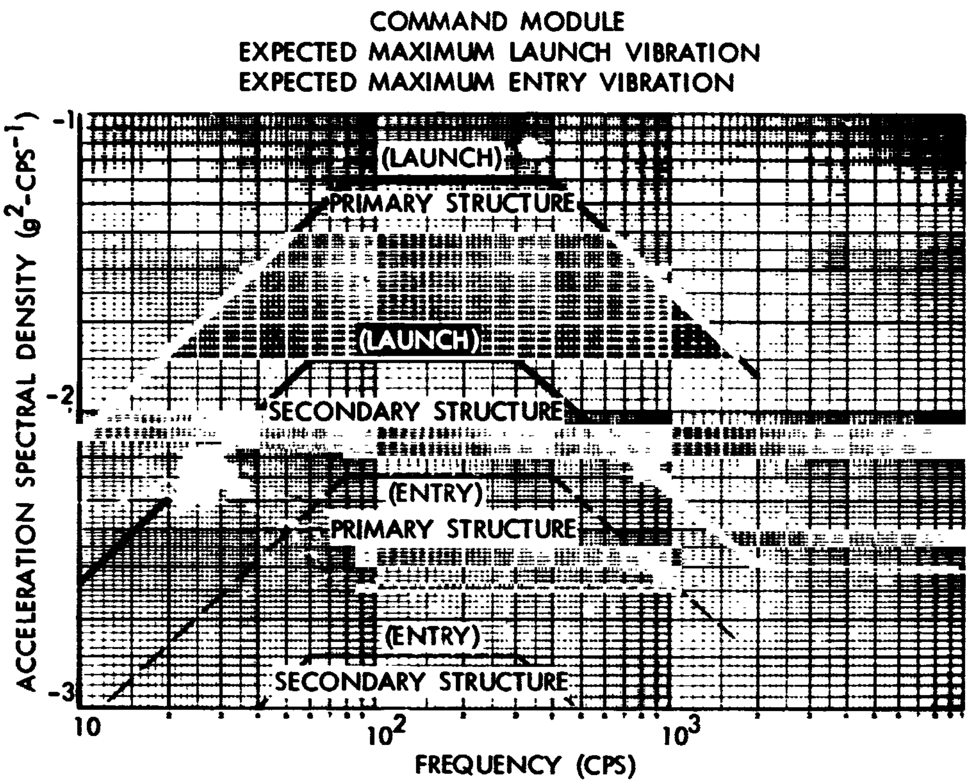


Figure 5. Maximum-Acceleration Spectral Densities on Apollo CM Structural Subsystems

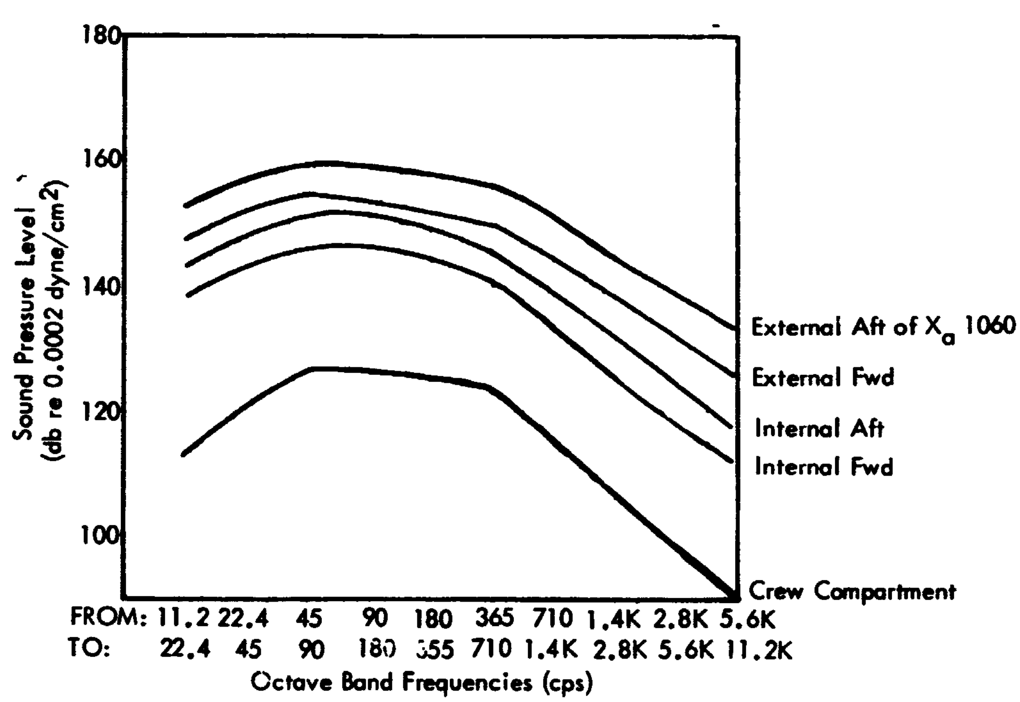


Figure 6. Apollo CM Launch Acoustical Levels

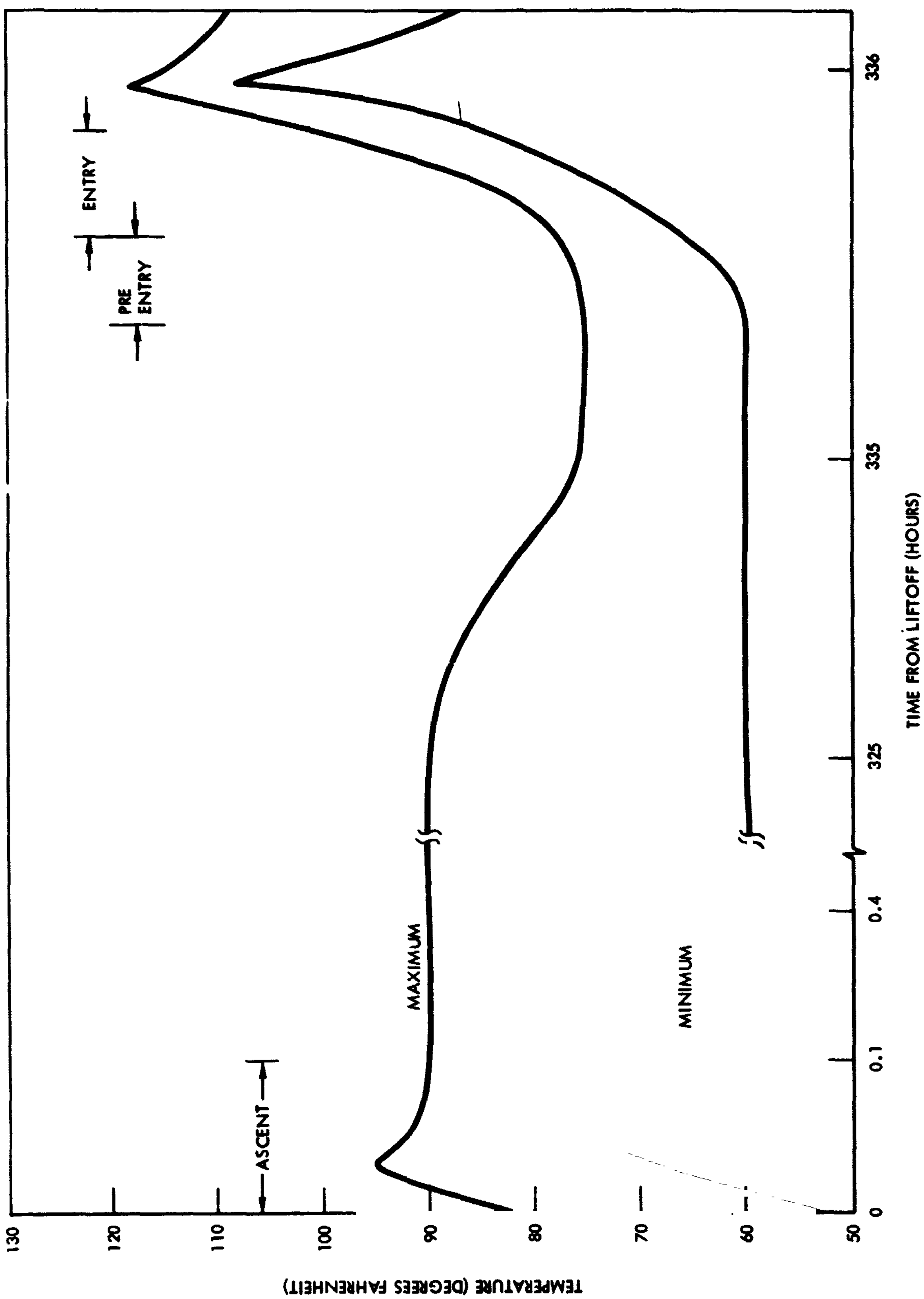


Figure 7. Average Temperature of CM Atmosphere

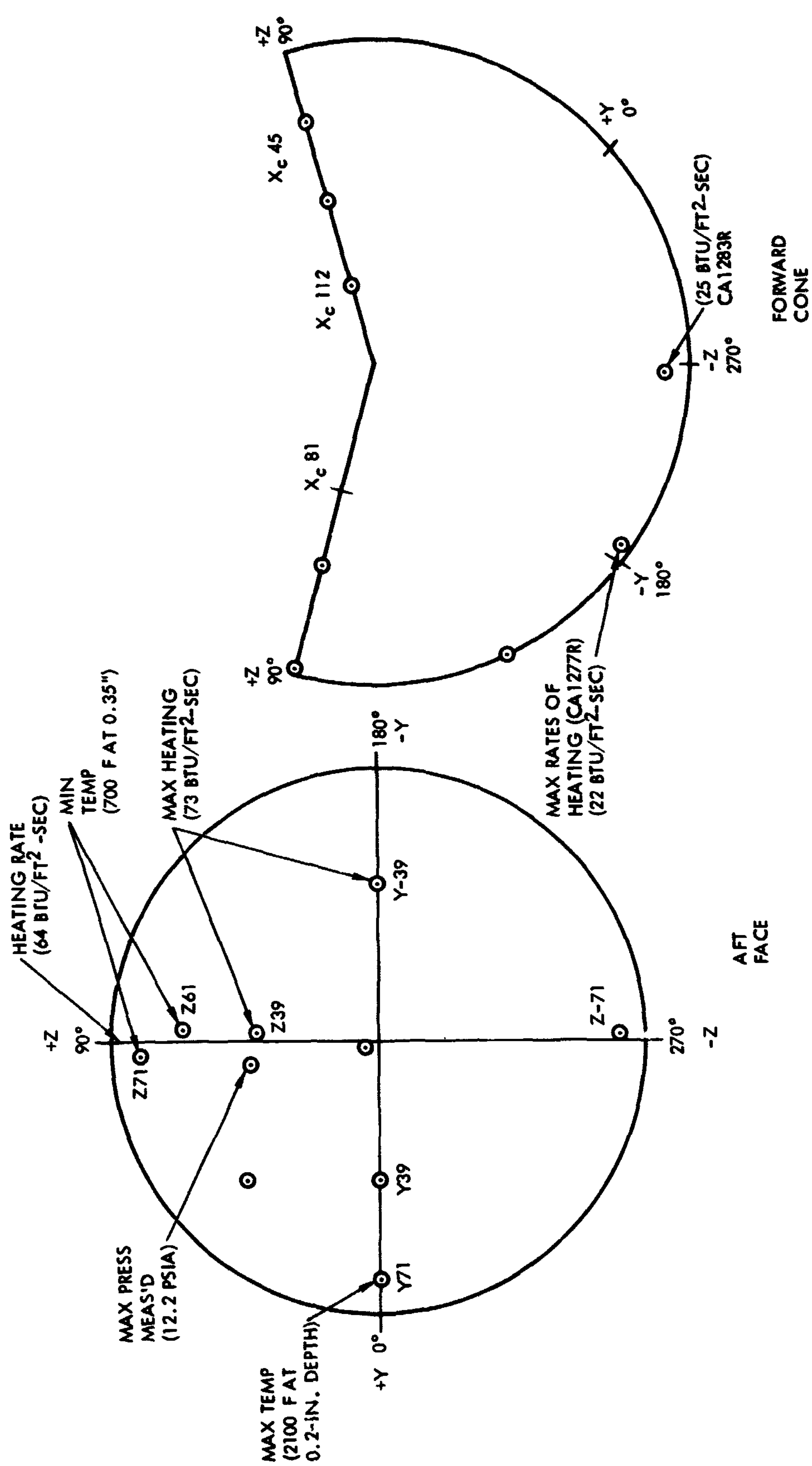


Figure 8. Apollo Heat Shield Measurement Points and Maximum Conditions

Actual maximum heating rates expected are between 160 and 190 Btu/ft^2/sec. On S/C 009, the three tension-tie rods were unsealed (the RTV sealant was omitted) and they melted flush with the fiberglass pad surface. Scorching extended through to the substructure around these regions.

At location Z71, at the stagnation point, for S/C 009, the temperature, char depth, discoloration, and heating rates exceeded predictions based on wind tunnel data. Other points indicated conservative design. Table 7 presents the predicted and S/C 009 observations.

Peak pressures and heating rates on S/C 009 were not observed due to the power drop at 1635 (70 seconds after entry). Prior data indicated that the peak values were within those expected, except for the stagnation point. Peak heating rate at the stagnation point is 164 Btu/ft^2/sec compared to the laminar theory (Riddell) prediction of 188 to 300 Btu/ft^2/sec.

Pressure time history for the command module is taken at 36 points on the heat shield. All valid S/C 009 pressure measurements were surveyed for maximum conditions. A maximum of 12.5 psia was observed. The preceding pressure agrees to 1 percent with the predictions for stagnation pressure based on equilibrium real air behind a normal shock and wind tunnel data at Mach 10, and attack angle of 22 degrees. Peak entry dynamic pressure was 930 pounds per square foot or 10 percent less than planned.

Table 7. Heat Shield Performance (S/C 009)

Location		600 F Depth		Char Depth (1000 F)		Surface Recession	
Z	Y	Predicted (in.)	Measured (in.)	Predicted (in.)	Measured (in.)	Predicted (in.)	Measured (in.)
71	0	0.39	0.47	0.25	0.32	0.08	0.06
0	0	0.42	0.35	0.29	0.21	0.11	0.02
-71	0	0.38	0.35	0.26	0.21	0.08	0.19
0	39	0.45	0.37	0.32	0.22	0.13	0.02
0	71	0.45	0.38	0.32	0.25	0.13	0.04

Structural Environment

Stress. Forward longeron and aft heat shield stress, and percent of maximum allowable loads are presented in Tables 7.1-II and 7.1-VII of Reference 4. All loads were well within tolerance, indicating more than

adequate design. The CM/SM tension tie rod stress measured not more than 41 percent full load allowable as an example of the point of greatest stress on the command module.

Dynamics. Detailed data are presented in Chapter 7 of Reference 4 for command module dynamics observations from S/C 009, showing that the acceleration spectral density exceeded that expected at 1200 cps for a dynamic pressure of 800 pounds per square foot at Mach 2.1 for points on the inner structure for S/C 009. (During supersonic flight, the spectral density at 250 cps was four times that expected for 790 pounds per square foot pressure at a location on the SLA near the RCM point of attachment).

Environmental Control Subsystem

Critical Temperature Profiles. ECS operation is critical to temperature of the water-glycol outlet-coolant circuit evaporator, glycol inlet, and cabin temperature. The ECS temperatures are presented in Table 8.

Table 8. ECS Temperatures

Parameter	Permitted Range		S/C 009 Observed	
	Min (°F)	Max (°F)	Min (°F)	Max (°F)
Water-glycol:				
Evaporator outlet	42°F	48	26.4	57
Cabin radiator outlet	70°F	80	52	67

Temperature extremes noted are indicative of problems other than externally induced environmental extremes.

Pressure Profiles. Cabin pressure regulation on S/C 009 was 5.65 psia, as compared to the 5±0.2 psia specified. Following touchdown, an amount of salt water leaked in through the steam duct outlet part and cabin pressure relief valve.

Spacecraft Windows. Contamination of windows appeared on landing or on entry; however, these windows should be acceptable for reuse after cleaning.

RCS System (CM)

The CM-RCS operated successfully up to electrical system failure at 1641 seconds after start. The CM-RCS temperature and pressure data for the S/C 009 flight are presented in Table 9.

Table 9. S/C 009 Temperature and Pressure

Tank Temperature (°F)			Tank Pressures (psia)		
Helium	Fuel	Oxidizer	Helium	Fuel	Oxidizer
60-69	63-65	63-65	3478-4104	66-285	61-285

CM Structure Temperatures at Touchdown

The following data was derived by analysis, and provides environmental conditions in which components and subsystems within the CM must survive and perform properly. Consequently, the severest of most probable environmental conditions were incorporated into the analyses to generate conservative answers and provide for undefinable variables.

Figures 9 and 10 present temperatures of the cabin structure at touchdown for a maximum heat load trajectory. The mean structure temperature at beginning of entry was 82 F and the entry period was 0.42 hours. The mean structure temperature at touchdown was 105 F. The temperature levels shown are the maximum values reached and will continue for a period of 1 hour after touchdown for a land landing; for a water landing, the period will vary from 15 minutes to 1 hour, depending on the time the hatch is opened and turbulence of the water.

Temperatures in the aft equipment bays of the toroidal section between the heat shield and the cabin varied from 150 to 160 F at touchdown and could increase to 200 F after touchdown because of soakback. Again, this soakback temperature and period will vary with the type of landing. These temperatures are based on the current heat shield design which is to be revised as a result of flight data. Interpretation of flight data and recalculation of heating rates will permit a thinner, lower-weight heat shield. The thinner heat shield will result in less virgin material remaining during reentry and, consequently, higher temperatures in the aft equipment bays. Therefore, the values presented should, at best, represent nominal temperature levels and not maximum values.

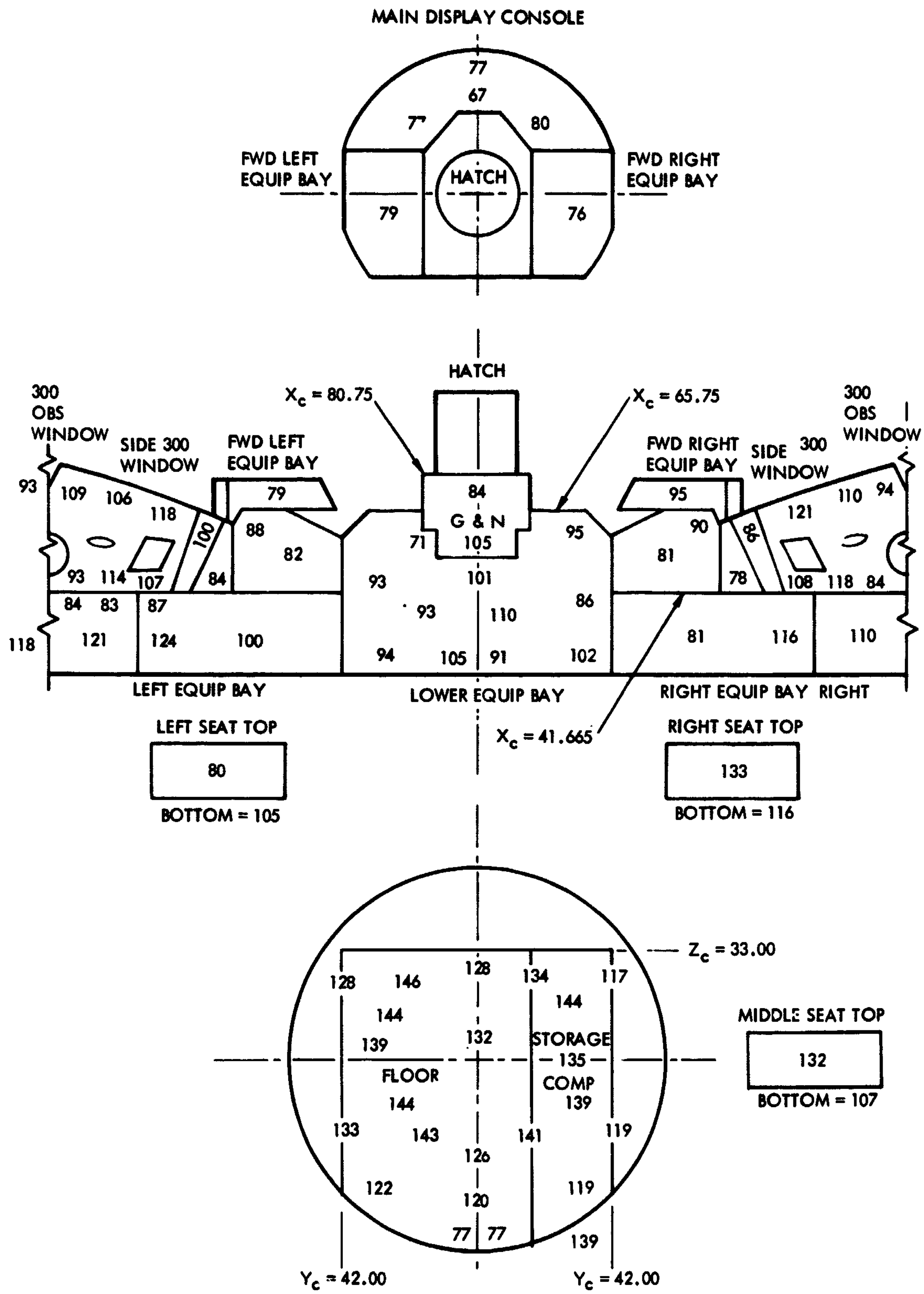


Figure 9. CM Cabin Structure Temperatures at Touchdown, Lunar Mission, Structure Exposed to Cabin Air

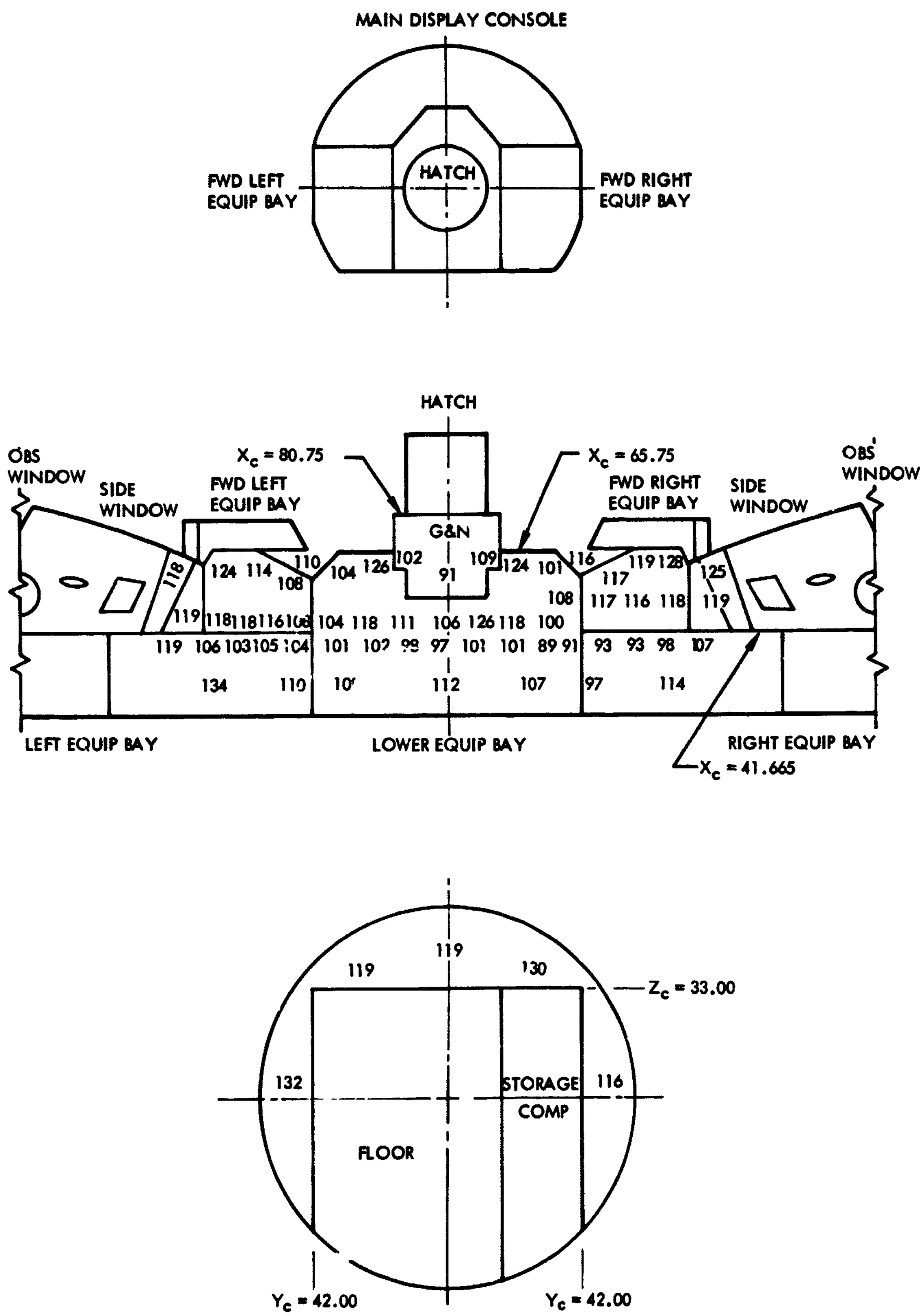


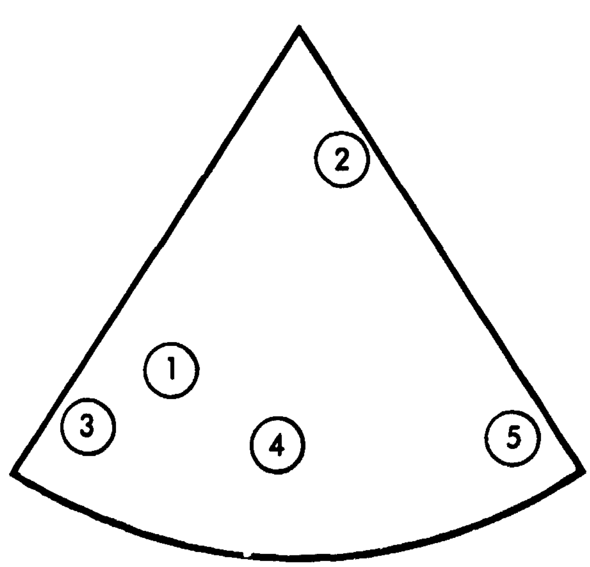
Figure 10. CM Cabin Structure Temperatures at Touchdown, Lunar Mission, Structure Not Exposed to Cabin Air

Apollo CM Subsystems Space Radiation

The Apollo command module subsystems will be subject to solar particle events, the solar wind, geomagnetically trapped particles, and secondary radiations. The significant dose contributions to certain important interior regions of the Apollo command module during the 14 day lunar mission are given in Table 10. The values given in the table were previously reported in Reference 5. The general locations of the dose points are given in Figure 11. The solar wind, which is encountered outside the geomagneto-

Table 10. Doses (Rad) to Interior Regions of Apollo CM From Fourteen-Day Lunar Mission (Reference 5)

Dose Component	Solar Protons and Alphas		Trapped Particles	Total Dose	
Probability of Encounter (Percent)	1	0.1	100	1	0.1
Region					
1. Lower equipment bay	6.24	59.3	0.079	6.32	59.4
2. Parachute compartment	51.1	487.0	0.37	51.5	487
3. RCS fuel tank	36.1	344.0	0.27	36.4	344
4. RCS oxidizer tank	8.94	85.2	0.11	9.05	85.3
5. RCS engine valve seat	55.7	531.0	0.40	56.1	531



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1. Lower Equipment Bay
2. Parachute Compartment
3. RCS Fuel Tank
4. RCS Oxidizer Tank
5. RCS Engine Valve Seat

Figure 11. Locations of Regions in Apollo CM at Which Doses are Evaluated

sphere, produces a maximum time integrated flux of 4.8×10^{15} proton-cm^{-2} with energies between 0.3 and 10 kev.

Trajectories and instrumentation of S/C 009 and 011 were not intended to provide correlation radiation levels with those predicted for the Apollo lunar mission. Ther . no immediate co elation of this data based on the results of these suborbital missions is forthcoming.

Observations of Apollo Environment (S/C 009 and 011)

The natural and induced environment experienced by S/C 009 were basically in all respects to the projected environment. The most important deviations were a result of the lower-than-planned entry velocity and lower resultant heating. No subsystems except possibly the ECS subsystem have been found significantly marginal in ability to perform the objectives of the missions. The operational period before malfunction expected for the ECS subsystem may be exceeded on the launch pad as was the case in S/C 011.

Spacecraft Flight Trajectory Parameters

Figure 12 presents the inertial and relative velocity profile for the S/C 009 mission, indicating that mission phasing can be accomplished close to schedule. The inertial flight path angle was as programmed within 2 degrees; longitude and latitude was within 0.5 percent of planned trajectory. Booster burning cutoffs were different than planned, thus a given altitude was reached later than expected and maximum entry velocity was one part in 27 less than planned.

Dynamic pressure, maximum g load, and Mach number is presented in Figure 13 for the entry portion of the S/C 009 flight. Maximum heating rates for the flight were less than expected, at least partially, as a result of the reduced entry velocity. The maximum heating rate for S/C 009 was 164 Btu/ft^2/sec.

Acceleration spectral density curves obtained from S/C 009 flights were all within expected levels during launch, except for measurement number AK025ID for vibration on the SLA skin radial panel, which is near the point where the RCM lab is to be carried. Vibration level at 250 cps at this point was found to be three times the expected level (specification or 2 g/cps. Maximum load factor on S/C 009 was 14.3 g.

Figure 14 shows CM/SM fairing temperature to be well below anticipated levels during the launch period.

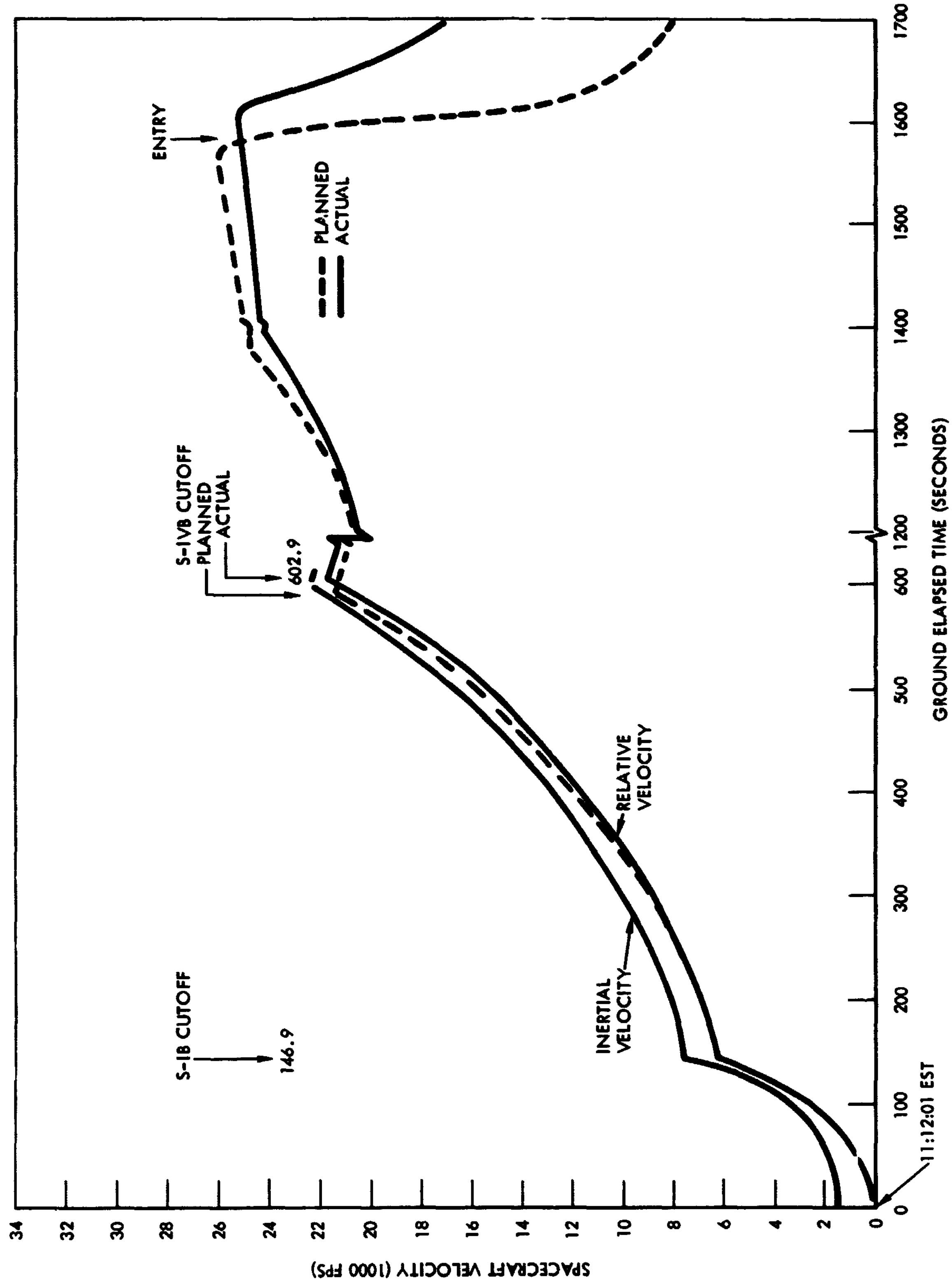
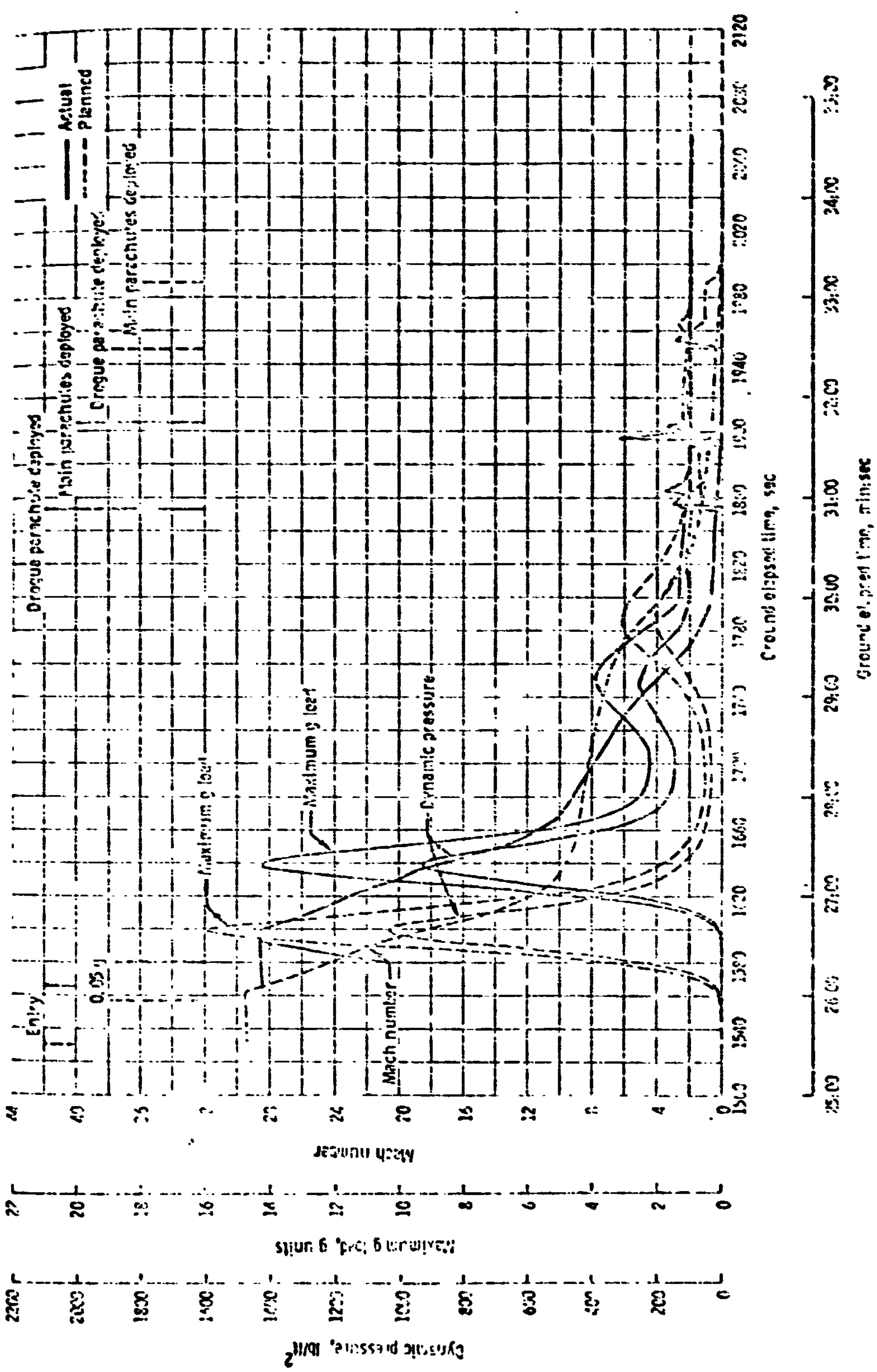


Figure 12. Spacecraft 009 Relative and Inertial Velocity Profiles



(a) Mach number, maximum g load and dynamic pressure.

Figure 13. Spacecraft 009 Mach Number, Maximum G Load, and

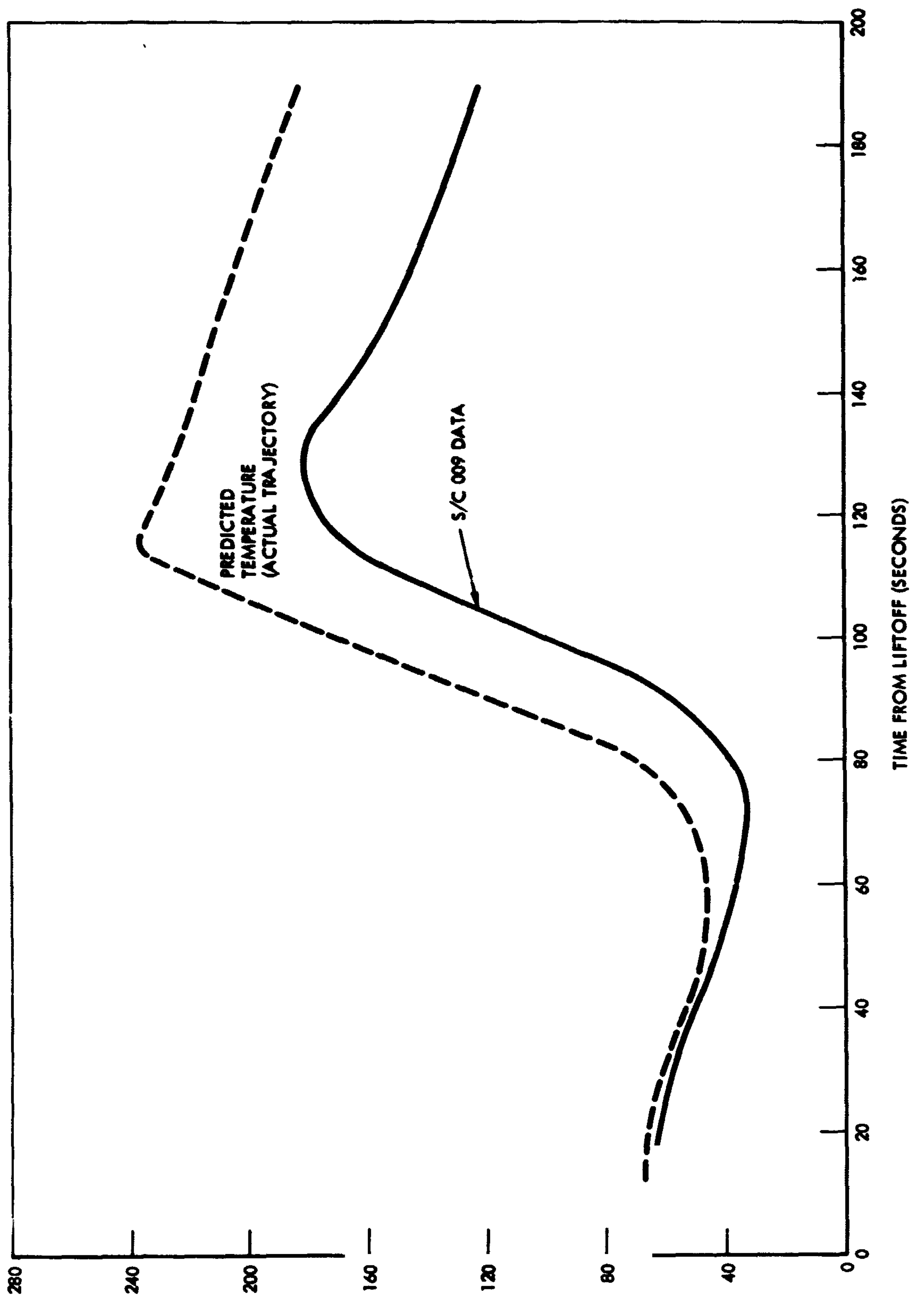


Figure 14. Temperature Versus Time for Outer Surface of CM/SM Fairing

Preliminary Data From Apollo S/C 011 Flight

The major objective for both S/C 009 and 011 flights was to qualify the heat shield for manned flight. Both flights were suborbital but with entry velocity increased above normal orbital return entry velocities (25,400 feet per second for S/C 009 and about 28,000 feet per second for S/C 011). Although S/C 011 entered with a higher velocity, a shallower entry angle was used so the spacecraft would skip back out and cool somewhat before final entry. The peak heating rate and maximum temperature was less, therefore, for S/C 011 than for S/C 009, however, the heat soak was longer with correspondingly higher temperatures on the pressure vessel structure. NASA data analysis indicated that S/C 011 met all major objectives for the flight, with high heating rates about one-half those experienced on S/C 009.

Tables 11 and 12 include results of preliminary quick look analyses of telemetry data from KSC, Antigua, and Canarvon.

Table 11. Selected Temperatures From S/C 011

Subsystem Area	Temperatures (°F)			
SM to LEM Adapter	Min	Time	Max	Time
SLA outer panel			400 (may be sensor error)	During boost
CM-PCM	89	Boost	90	Boost
S-band power amplifier	110	Boost	111	Boost
VHF/AM XMTR	104	Boost	106	Boost
S-band transponder		Boost	110	Boost
C-band transponder			105	Boost
Collins signal conditioner			90	Boost
ECS space radiator outlet			49	Boost
ECS glycol evaporator	46		50	T + 120 sec
outlet (F 0018)			53	T + 200
	35	T + 500 sec		
	40	T + 550 sec		
(F 0017)	40	T + 100 sec	77	Launch
	30	T + 500 sec		
Suit supply	56			
Cabin	75			
Static inverter				
(C 0179)	79	Boost	80	Boost
(C 0178)		Boost	105	Boost
(C 0177)	80	Boost	90	Boost
(C 0176)		Boost	107	Boost
(C 0175)		Boost	140	Boost

Table 12. Selected Pressures From S/C 011

Subsystem Area	Pressure	Time
B fuel manifold	200 psia	Boost
B oxidizer manifold	185	Boost
A fuel manifold	190	Boost
A oxidizer manifold	190	Boost
Cabin pressure	16 psia	On pad T = 0
	6 psia	at T + 150 sec
Baro static reference	14.15 psia	On pad T = 0
	0.0 psia	at T + 130
	(then went negative)	

Miscellaneous comments regarding the Apollo S/C 011 flight are the following:

1. Tension tie maximum force density = 8 - 11 x 10^3 psi
2. Landing struts showed no stroking
3. Electrical loads were 20 percent lower than expected
4. Communications were all normal and complete
5. On-board tape sync problems can be corrected
6. EDS-bending moments on SM/CM were normal
7. CM windows were clear up to entry, but right-hand window became obscured by condensation between the heat shield and pressure vessel region of depth after entry. Movies show a cherry-red ablator on windows during entry.

S/C 009 Failure and Stress Items Reported

Report No.	Measurement No.	Remark
SID 66-487		CM propellant tank bladder leaks, 250 cc/3.5 sec (130/15 min ok).
SID 66-485	C 14A7, 8	Open circuited.

Report No.	Measurement No.	Remarks
SID 66-519		HF recovery beacon inoperative after impact (part did not fail, but possible power problem).
SID 66-520		HF recovery beacon inoperative after impact (part did not fail, but possibly power problem)
SID 66-539	CA 0415S	Strain sensor failed at T + 183 sec (problem external to sensor)
SID 66-540	CE 0035P	Pressure sensor shorted with 17.0 volts out (Zener diode failure)
SID 66-563		Low range acoustic transducer failure due to mishandling prior to flight (may be indicative of over voltage)
SID 66-564	CA 7650P	Pressure-shift in calibration by 2%
SID 66-576	CA 1149P	Pressure (heat shield) calibration shift (transducer problem)
SID 66-576	CA 1172P	Pressure (heat shield) calibration shift (transducer problem)
SID 66-576	CA 1177P	Pressure (heat shield) calibration shift (transducer problem)
SID 66-588		System A fuel helium relief valve burst disc ruptured (design pb)
SID 66-601		CM RCS engine valve degradation effects normal
SID 66-619		HF beacon failure due to salt water - reduction of radiated power
SID 66-620		VHF/AM transmitter modulation wave-form distortion (ok)

Report No.	Measurement No.	Remarks
SID 66-638	All heat shield Instruments	Thermocouples inoperate after T + 1625 sec (power failure). Char on aft heat shield below 0.2 inch (sea level). Heat shield saturated with water and salt (conductive!). (Still drying 30 days after flight). Salt water intrusion renders sensors inoperative without drying (electrically ok but calibration shifts). Some 5% en/26% W thermocouples shorted at 1000°-1400 F. At stagnation point, heat shield backface reached 180-200 F.
SID 66-687		CM RCS propellant solenoid valve (oxidizer valius) leaking 50 cc/30 min (failure due to causes during flight or after landing)
SID 66-696		V16-327280 bolt assembly damaged in flight but not due to tension-tie explosive separation.
SID 66-699		Windows contaminated with sodium, etc. (salt water and dirt) but part deposited in flight (20% transmittance).
SID 66-710		CO_2 and Freon mono-F in CM atmos ok but 300 ppm methyl alcohol toxic with 14 day exposure when found at Downey (hatch had been opened already). Traces of fuel and Teflon should have been present.
SID 66-711		Water-Glycol pH and NaMBT inhibitor ok. TEAP inhibitor 1.75% (1.6% ok) (possibly error in preparation). Particulate contamination 6 x allowable picked up from S/C plumbing.
SID 66-712		Waste water contamination minimal.
SID 66-713		Cabin pressurization leak 0.4 lb/hr to 14 lb/hr from flight data. (Press relief valve cracks open at 5.58 psi to 5.40 psi).

Report No.	Measurement No.	Remarks
SID 66-722		Side ablative hatch ground support type of gear had excessive rust and lead tape seal had large indentations resulting in 525 in. -pounds torque being needed instead of 100 to disengage hatch.
SID 66-723		Impact struts and ribs showed no stroking or damage. Uprighting system not even used and showed no damage. Crushable ribs showed no crushing.
SID 66-731		Thirty-seven of forty-five ordnance devices had fired properly; eight did not fire due to circuit short which tripped off the breaker.
SID 66-732		No structural failures or secondary bond failure. No evidence of structural failure in heat shield but extreme discoloration noted in each of three tension tie bolt holes in aft heat shield and some on inner structure aft longer on fittings. Excessive corrosion on main hatch mechanism pinion. Tear in right-hand drogue mortar can. Delamination and breakage in right-hand umbilical fitting.
SID 66-745		Heat shield instrumentation: pressure transducer shift in calibration; indications of transducer leakage into reference vacuum through diaphragm (header). (Fault has been avoided in later procurement).
SID 66-814		The CM/SM umbilical was burned and shorted during entry which caused numerous shorts and malfunctions to the CM (The SCS solenoid drivers may have been damaged).
SID 66-810		In 2-1/2 hours in the ocean, three pints of sea water got into the crew compartment through the steam vent and cabin pressure relief valve.

Report No.	Measurement No.	Remarks
SID 66-943		Umbilical short during entry caused overload on wire attached to C23 T B1-12, and breaker No. 18 supplying logic "B" power to MESC, CP; and PLSC blew at T + 1635.

III. STRUCTURES AND MATERIALS

This section covers the structures and materials renovation feasibility aspects of the Apollo CM as related to subsystems.

PRIMARY AND SECONDARY STRUCTURE

For the purpose of this study, the primary and secondary structures are considered as subsystems of the CM.

Repair Feasibility

Repair or replacement will be required for those structural components which were designed for entry heating or water impact, and were intended for a single flight mission. The structural components which fall into this category are:

- Boost cover
- Forward heat shield
- Aft heat shield
- Crew compartment heat shield
- Crew couch shock strut cores
- Shock attenuation panels
- Aft compartment crushable cores

Structural components for which the impact loads are critical but can be expected to tolerate this environment with little or no damage are:

- The command module inner structure
- The equipment support structure
- The crew couch and supporting struts

Repair and Replacement of Structural Components

Boost Cover and Forward Heat Shield

The boost cover is ejected following boost, and the forward heat shield is ejected before parachute deployment. These components, therefore, must be replaced for command module reuse.

Crew Compartment Heat Shield Substructure

Any damage the crew compartment heat shield subsubstructure will probably suffer at the time of impact can be repaired by standard repair methods. The structure should be inspected for weld cracks, damage to upper and lower rings, and overall distortion. X-ray inspection is required to detect core-to-facesheet voids in the steel honeycomb sandwich.

Aft Heat Shield Substructure

It is possible that the aft heat shield substructure may suffer extensive damage in the impact area which would involve failure of the steel facesheets and crushing of the honeycomb core. Damage of this nature will probably be concentrated in one 90-degree segment, and this can be repaired by cutting out the segment along the weld lines and replacing with a new panel. The remaining area is inspected for weld cracks, damage to the upper ring, and overall distortion. X-ray inspection is required to detect core-to-face sheet voids in the steel honeycomb sandwich.

Crew Couch Shock Strut Cores

If any stroking of the crew couch shock struts takes place during impact, the crushable cores must be replaced.

Shock Attenuation Panels

If a side load is experienced during impact, the shock attenuation panels may be damaged. Any damaged panels should be replaced.

Aft Compartment Crushable Cores

The aft compartment crushable cores are intended to protect the inner structure in the case of high-impact loads in this area. The condition of these cores can be determined by visual inspection and may be replaced as required.

The Command Module Inner Structure (Pressure Vessel)

The command module inner structure is expected to suffer little or no damage during impact, and any repairs required should be within the scope of current standard repair methods. To establish the structural integrity, however, a complete inspection will be required for bond de-laminations, core-to-face-sheet voids in the honeycomb panels, and cracks in the inner skin welds. Voids in the outer facesheet metal-to-metal bonds would indicate that the honeycomb core may have been exposed to sea water.

The Equipment Support Structure

Equipment support structure tees and brackets bonded on the inner surface of the aft bulkhead may experience damage at the instant of impact. These items can be removed and replaced by standard repair methods.

The Crew Couch and Supporting Struts

The crew couch and supporting struts will not suffer any damage during a normal impact.

FATIGUE ANALYSIS

A conservative and simplified fatigue analysis has been made for the purpose of showing that fatigue would not become a problem until the structure had been exposed to a large number of flight missions. The epoxy-phenolic adhesive used extensively on the inner structure is the material most sensitive to fatigue. The damping response of the structure has been considered conservatively to be low, and this consideration is reflected in the number of load applications used; i.e., a single load application at a high stress level has been taken 20 times by the structure before it is damped out. This analysis is presented only as an indication of the low level of fatigue damage expected (Table 13) and to demonstrate the acceptable nature of the existing 1.50 safety factor (Figure 15). All components exposed to significant stress levels during flight have been tested to the design ultimate load. Table 14 shows a list of such tests performed on various Apollo specimens.

CONDITION OF RECOVERED STRUCTURE

One of the basic objectives of the Apollo command module design was to provide a usable structure for a single mission. There is no structural significance to this requirement over the entire load spectrum of the mission until the instant of impact is reached. The most extreme loading that the vehicle can be exposed to during pre-launch handling, launch, boost, abort staging, docking, space flight, and entry will not exceed two-thirds of the material capability of all materials used in the structure. It follows that the structure may be subjected to all of the loads in this spectrum for a great many times until the point of material fatigue is reached, say after 100 missions. At the instant of impact some areas of the structure will be loaded to the ultimate capability of the material and, in extreme cases, to the point of failure. It is in these areas only that the single-flight design objective is structurally significant. Structural components that may experience ultimate loads at the instant of impact are the aft heat shield, the shock attenuation cores in the aft equipment compartment, the shock attenuation panels on the right- and left-hand lower equipment bays, tees and brackets bonded on the aft bulkhead of the inner structure in the impact area, and the couch attenuation strut cores.

Table 13. Cumulative Fatigue Damage

Condition	Ultimate Stress Level (percent)	Load n Applications	N	n/N
Ignition	67	20 †	20,000	0.001
Boost	67	15 †	20,000	0.00075
Max q α (Vibration)	67 5	20 5000	20,000 $(100)(10)^6$	0.001 0.00005
Staging	50	20 †	20,000 to 10	0.00002
Abort	67	0	$1(10)^6$	0.0005
Impact*	67 to 100	10 †	20,000 to 10	

*100 percent stress level applies to aft heat shield at the point of impact.

$\Sigma n/N = .00332$

N required for zero margin = 20,000 (.00332) = 66.4

Stress level at 66.4 cycles = 100%

Required factor $= \frac{100\%}{67\%} = 1.50$

NOTE: These values are approximate and are based on limited data. Their presentation is intended to indicate a degree of magnitude only, and to demonstrate that a vehicle subjected to the load spectrum of a single flight suffers no fatigue damage.

†The low damping response of the structure is conservatively assumed to result in the quoted number of cycles at full load.

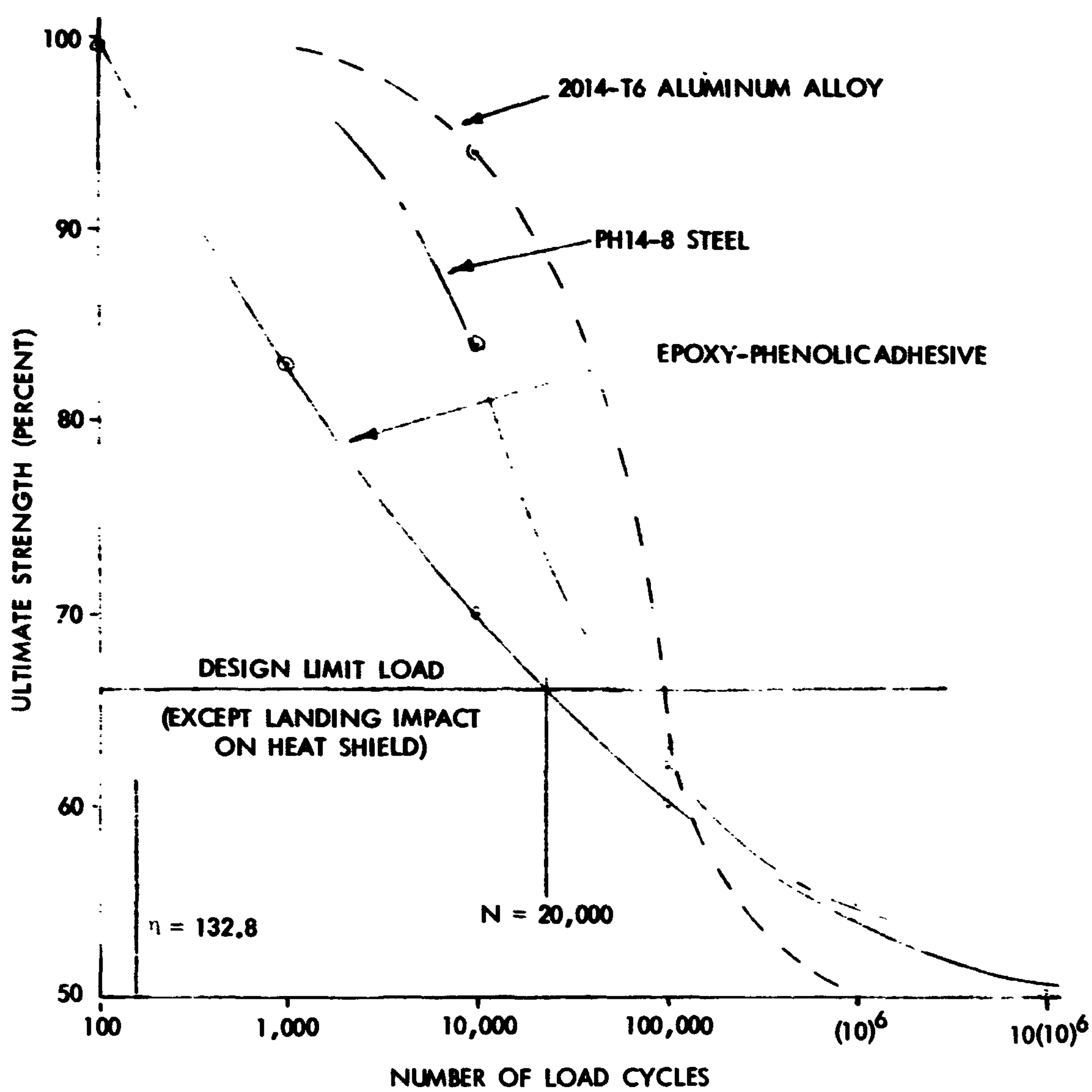


Figure 15. S/η Curves of Command Module Materials

Table 14. List of Tests Performed on Various Apollo Capsules

Specimens	Test	Remarks
004-A	Thermal	See TR 251003
004-A	Static Load	
004-A	Main chute deployment	See Note 1
004-A	Drogue chute deployment	Tested to ultimate
004-A	Crew couch attachment	Tested to ultimate
004-A	Pilot mortar	Tested to ultimate
004-A	Abort condition	Failed at 165% of limit
004	Left and right hand equipment	Tested to ultimate
004	Thruster load condition	Tested to ultimate
004	Drogue chute mortar	Tested to ultimate
004	Burst Pressure	12.9 psi for about two minutes
BP 28	Water Drop	
	Nos. 91, 92	See Note 2
	Nos. 97, 99, 101	See Note 3
007	Water Drop	
	Nos. 100, 102	Satisfactory
006	Vibration	See Note 4

Notes:

1. Three main parachute deployment tests were performed, each with the design-ultimate load modified by a temperature-correction factor. Said load was applied in three different directions: over the tunnel, straight out, and normal to the longeron. At the third test, failure occurred at 14 percent of limit. However, that was considered satisfactory because the temperature-correction factor assumed was conservative.

2. Boilerplate model, except for aft bulkhead, aft heat shield, and crushable ribs, which are production articles.

3. Same as Note 2, with plumbing, tanks, and crushable rib fittings installed.

4. The test performed is equivalent to a vibration exposure encountered in about 19 missions. However, the test module is significantly different from flight specimens as regards equipment installations, e.g., gross weight, etc.

Degradation of the structure that may be considered significant will result from exposure to the following environments: landing impact, post operational handling, and exposure to sea water and other corrosive media. Degradation due to landing impact will be local in area and subject to repair. Degradation due to post operation handling can be avoided. Degradation due to corrosion, if not prevented or controlled, could result in the total loss of the structural capability of the basic structure.

CORROSION INVESTIGATION

Postflight evaluation of the 009 and 011 CM crew compartment and heat shield structures and the metal and electrical components of the control and support systems indicated that corrosion degradation might be the primary limiting factor governing their potential structural and operational integrity.

The CM is weighted to be maintained in a vertical position subsequent to splashdown and until recovery aboard ship. Sea water enters through the heat shield hatch and side ports in the crew compartment heat shield and fills the space between the heat shield and the crew compartment to a level above the joining plane of the aft- and side-heat shields. The CM will remain in the ocean for an indeterminate period and additional time may expire before the space is drained. Currently no provisions have been made to drain the sea water or to flush the flooded area. The crew compartment, systems materials and equipment, and brazed heat shields thus are exposed to prolonged immersion in the corrosive sea water media.

The potential-reduced reliability of the systems, the potential degradation of the mechanical properties of the crew compartment structural materials, and the potential structural damage to the brazed heat shields are serious considerations in this refurbished CM study. Two comparison laboratory studies were therefore conducted: the first, to evaluate the potential effect of corrosive damage on the mechanical properties of the core and facing sheet; and the second, to determine the feasibility of flushing the assembly and neutralizing the corrosive effect of the sea water.

EFFECT OF CORROSION ON MATERIAL PROPERTIES

Major consideration with respect to the reuse of the structure and components of Apollo command modules has been directed toward corrosion prevention. Material selections for Apollo components and the use of special protective coatings (per NAA Engineering Drawing V17-000024) precludes or retards corrosion. Nevertheless, corrosion deterioration can occur to certain components, assemblies, and structures during flight and by exposure to sea water after earth reentry.

Aft Brazed Heat Shield Constructions

The aft brazed heat shield is the structure most vulnerable to sea-water corrosion. The general construction of the heat shield and its relative position with respect to the crew compartment structure is illustrated in Figure 16. The ablative material is an epoxy-phenolic resin formulation reinforced and lightened with strategic fillers and a fiberglass honeycomb matrix.

The face sheets and honeycomb core of the heat shield structure are PH14-8, a precipitation, hardenable stainless steel. The PH14-8MO material was selected because it affords high strength and corrosion resistance. The corrosion resistance of PH14-8MO is intermediate between that of the 18-8 grades of stainless steel and the lesser corrosion resistant 12 chromium grades, such as type 410 stainless steel. Adherence of the face sheets to the honeycomb core is achieved by furnace brazing, Figure 17. The braze alloy used is the LTCM alloy, a silver braze alloy with nickel filler (20 percent by volume). The Apollo honeycomb braze detail is illustrated in Figure 17.

The core material of the assemblies proposed for refurbishment contains perforations of 0.008 inch diameter spaced at 0.375 inches O.C.

Brazed fillets are formed at the top and bottom face sheet to honeycomb core interfaces and a thin film braze deposit is adhered to much of the honeycomb core cell walls during the braze cycle and heat treatment.

Evaluation of Postflight Hardware

The S/C 009 aft heat shield was not available for critical examination when the RCM study contract was initiated. A detailed evaluation of the aft heat shield from S/C 011 was therefore undertaken to determine the effect of extended contact with sea water on the quality and strength of the core and braze attachment.

The Apollo S/C 011 CM flew and was recovered 25 August 1966. The weight of the command module had increased 560 pounds. This is attributed to its absorption and entrapment of sea water. The aft heat shield was removed 16 September 1966. Segments for thermal analyses were removed from the aft heat shield, starting 24 September 1966. Segments for the corrosion evaluation were removed, at the locations designated in Figure 18, between 1 and 5 October 1966.

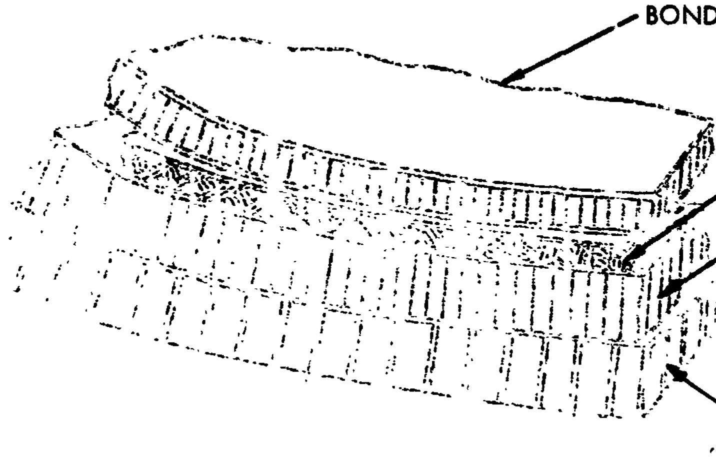


Figure 16. Basic Apollo Command Module Structure

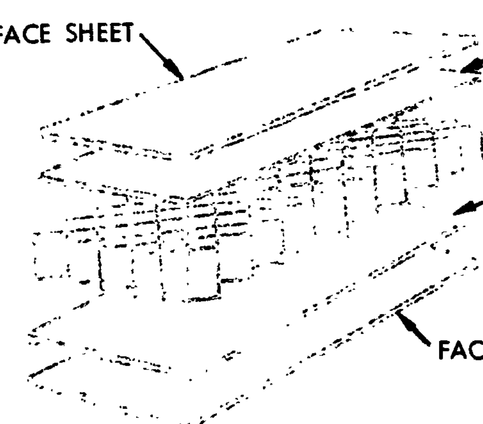
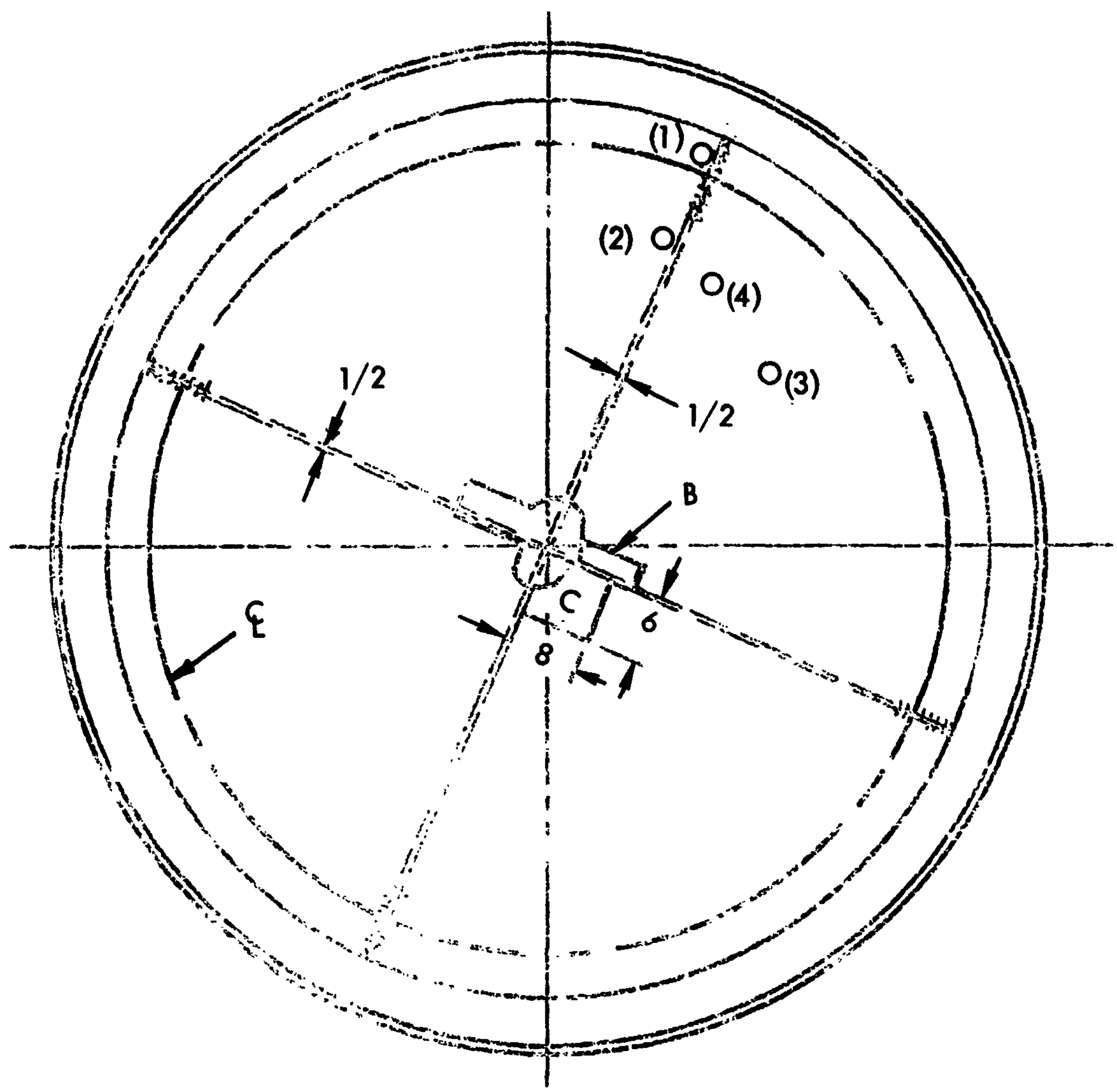


Figure 17. Brazed Honeycomb Sandwich Detail



SPECIMENS 1 AND 2 - PLUGS CUT ALONG CORE SLOT

SPECIMENS 3 AND 4 - PLUGS CUT IN OPEN FIELD OF CORE

SPECIMEN C - CENTER CORRODED SECTION

SPECIMEN B - INNER SKIN PEELED BACK

Figure 18. Corrosion Study Specimen Locations

A visual examination was made of the representative core segments and several facts became immediately apparent.

1. The core foil in all of the plug specimens, including all but two of those removed for thermal evaluation, did not contain perforations.

2. Corrosion within the brazed assembly was limited to surfaces which had been in direct contact with the entrapped sea water. The sea-water contact was limited to the outermost nodes and cells of the PH14-8MO core. The corrosion was identified by discoloration (brownish-red) and minute pitting.

3. The central section of the assembly, which still contained entrapped sea water 20 days after splashdown and moisture after 30 days, showed the greatest discoloration and pitting.

4. The unexposed cells evidenced no corrosion.

5. The heat shield construction was not representative of the later Block I and Block II assemblies proposed for refurbishment.

<u>Laboratory Test Program</u>

When it was determined that the postflight evaluation of the S/C 011 heat shield would not be representative of the proposed refurbished assemblies, an alternate test program was instituted.

Specimen Preparation

A PH14-8MO honeycomb sample 2-1/2 in. thick by 3-1/2 by 9 in. was taken from the trimmed area of an early nonperforated-core heat shield configuration and prepared for test as follows:

1. The test panel was cut into six pieces 2-1/2 in. thick by 3-1/2 by 1-1/2 in., and one facing of each segment was removed.

2. All of the pieces, including the removed facings, were saved for immersion tests.

Exposure Conditions

All tests were conducted in a saline test solution (ASTM D1141) with a combination of chlorides representative of sea water. Local sea water was not used because of unknown contaminations imposed by sewage and commercial effluents.

Four of the segments were submerged to one-half their depth, and the cells filled to a like depth, in the test solution. One segment, for each test condition, was removed and the entrapped fluid dumped after 1 hour, 3 hours, 24 hours, and 74 hours.

Two unexposed segments were saved for control standards.

Specimen Evaluation

Photographs were taken of the control and exposed test specimens. The control specimen is shown in Figure 19. The specimens exposed in the test solution for one hour and for three hours are also represented by Figure 20, since there was no visual evidence of corrosion. The specimen exposed for 24 hours is shown in Figure 21. The specimen exposed for seventy hours is shown in Figure 22.

Sections of the specimens from the area where the meniscus flow of the test solution extended up the cell wall (above the fluid level) were removed and mounted for microscopic examination. Sections were also taken from the face-to-core brazed joints that had been completely submerged for the various exposure periods. The pertinent micro and macro photos are depicted in Figures 23 through 30. The pertinent reproduction data are noted.

Conclusions and Recommendations

The following general conclusions and recommendations can be drawn as a result of these tests fortified by additional test data from Armco Steel Corporation, the developers of the PH14-8Mo alloy.

1. The control specimen and those exposed for one and three hours show no evidence of corrosion products. Therefore, it is concluded that parts adequately flushed and neutralized after these periods of sea-water exposure are not deleteriously affected.

2. The specimens exposed for twenty-four hours showed no evidence of intergranular corrosion but did evidence presence of corrosion products as identified by local discoloration. If adequately flushed and neutralized, the parts could be reused with structural reliability.

3. The specimens exposed for seventy hours showed no distinguishable intergranular corrosion. However, the corrosion products were noticeably increased as evidenced by local discoloration. Also, there was visual evidence of minor local pitting which had not been discernable in the particular photomicrograph made of that area of the specimen.

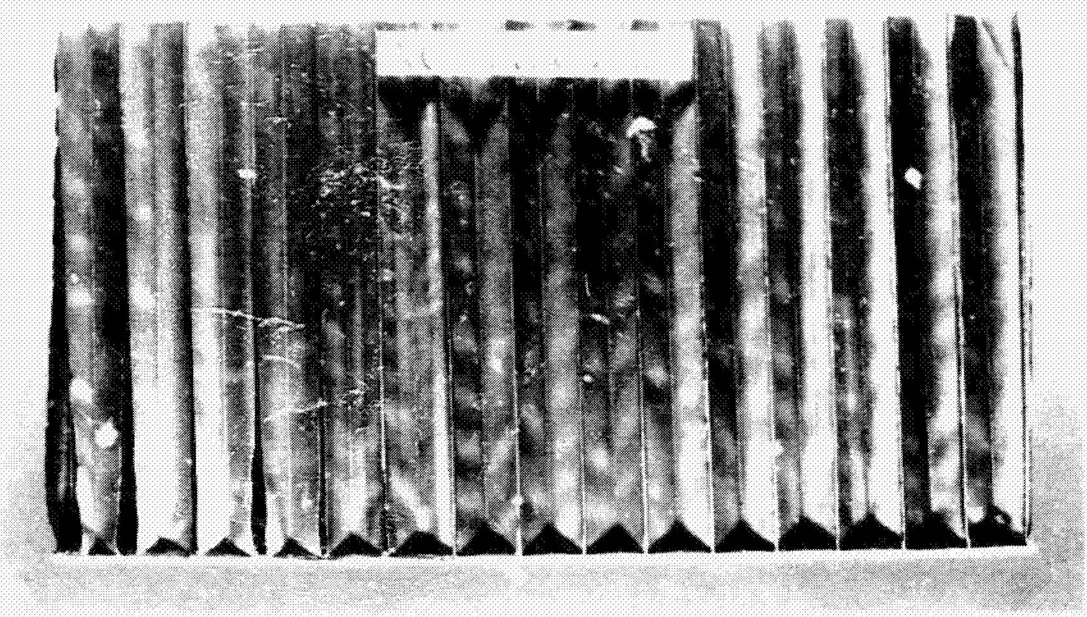


Figure 19. Laboratory Immersion Testing of PH14-8 Honeycomb, Control Specimen, One-Hour and Three-Hour Exposures

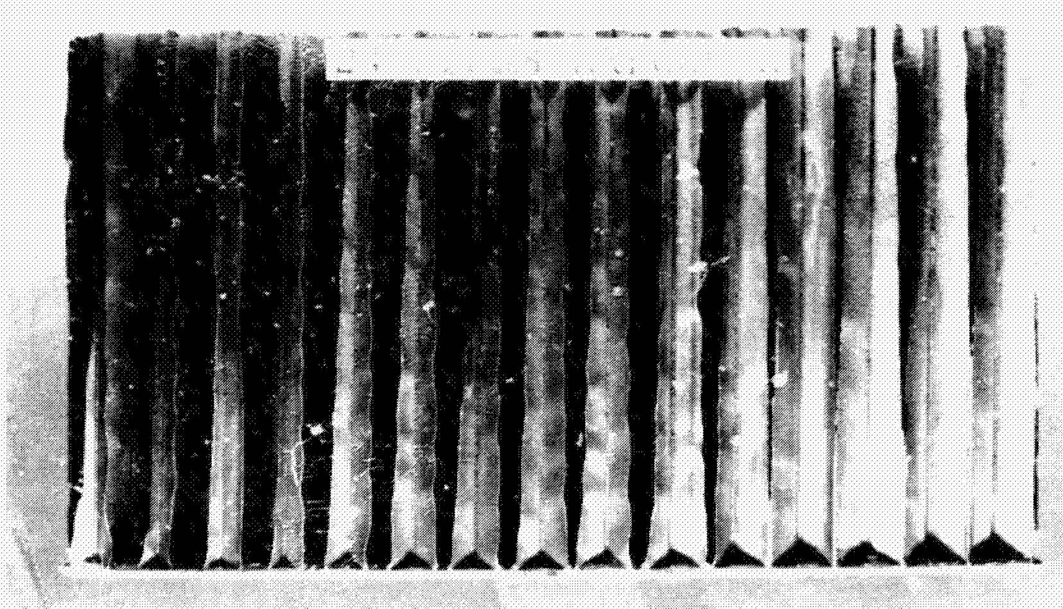


Figure 20. Laboratory Immersion Testing of PH14-8 Honeycomb, Twenty-Four-Hour Exposure

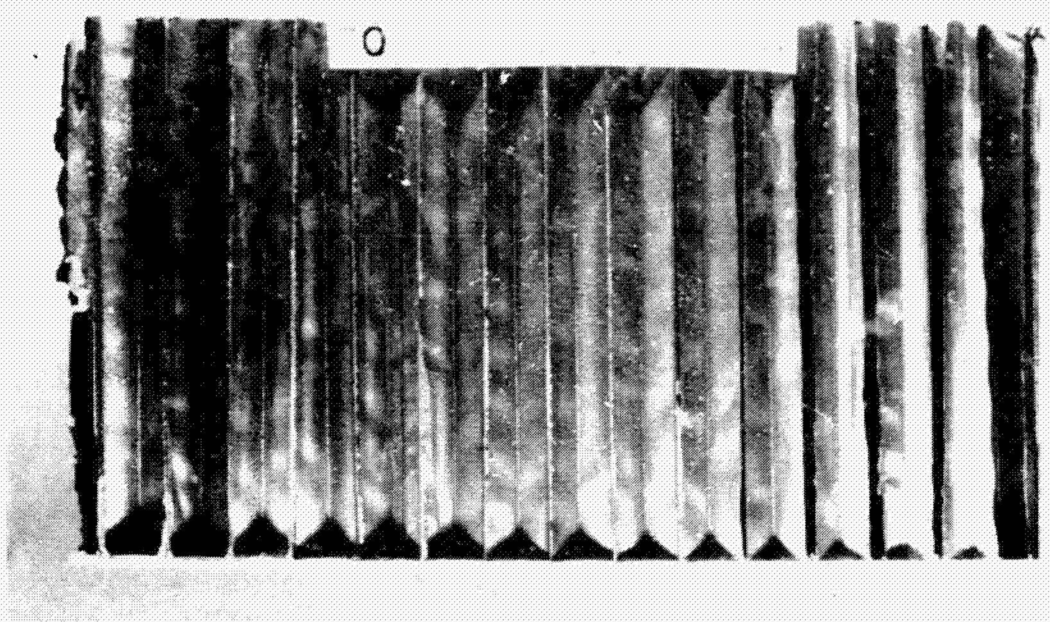


Figure 21. Laboratory Immersion Testing of PH14-8 Honeycomb, Seventy-Hour Exposure

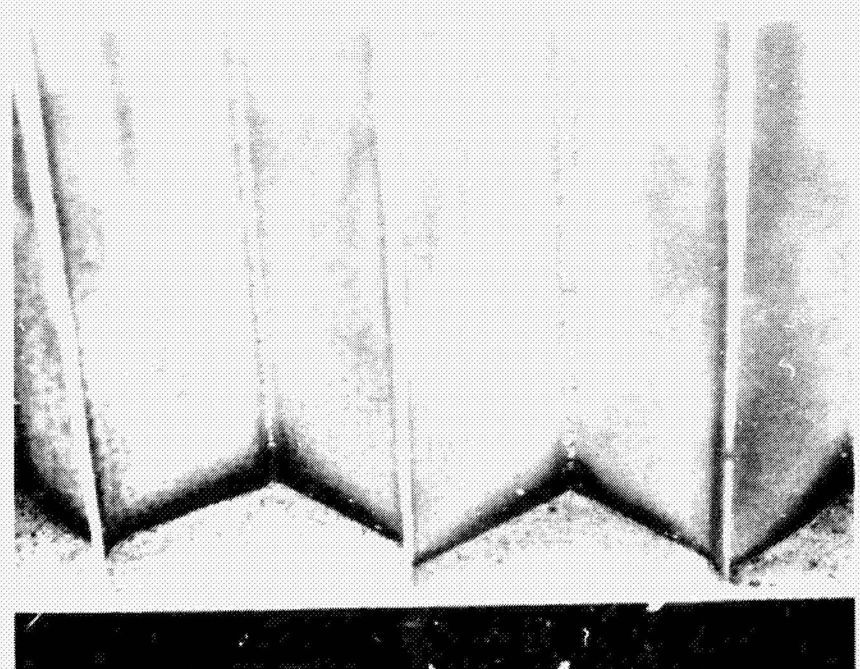


Figure 22. Laboratory Immersion Testing of PH14-8 Honeycomb, Seventy-Two-Hour Exposure, Magnification 6.6X

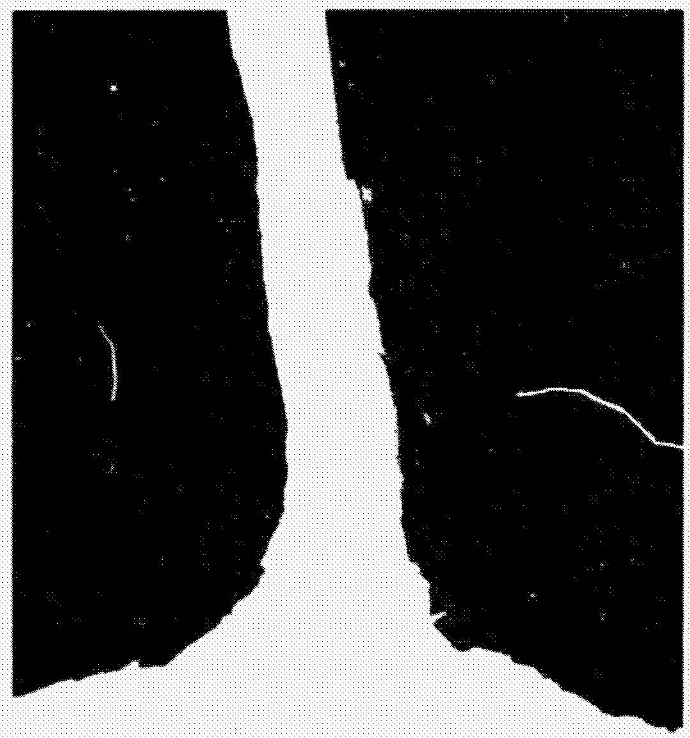


Figure 23. Laboratory Immersion Testing of PH14-8 Honeycomb, Control Specimen, Face-to-Face Core Braze, Magnification 200X

Figure 24. Laboratory Immersion Testing of PH14-8 Honeycomb, Control Specimen Section Through Core, Magnification 200X

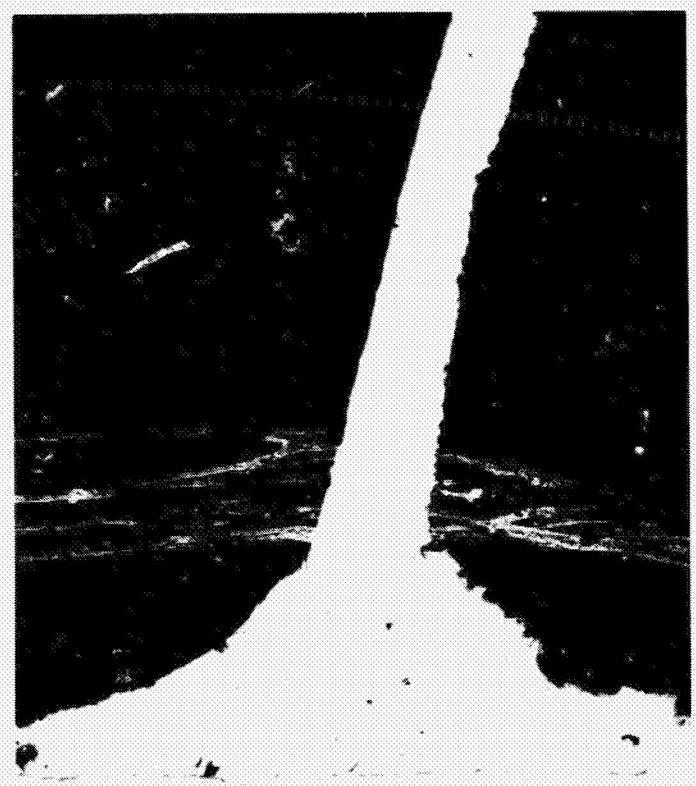


Figure 25. Laboratory Immersion Testing of PH14-8 Honeycomb, Three- Hour-Exposure Specimen, Face-to-Face Core Braze, Magnification 200X

Figure 26. Laboratory Immersion Testing of PH14-8 Honeycomb, Three- Hour-Exposure Specimen, Section at Meniscus, Magnification 200X

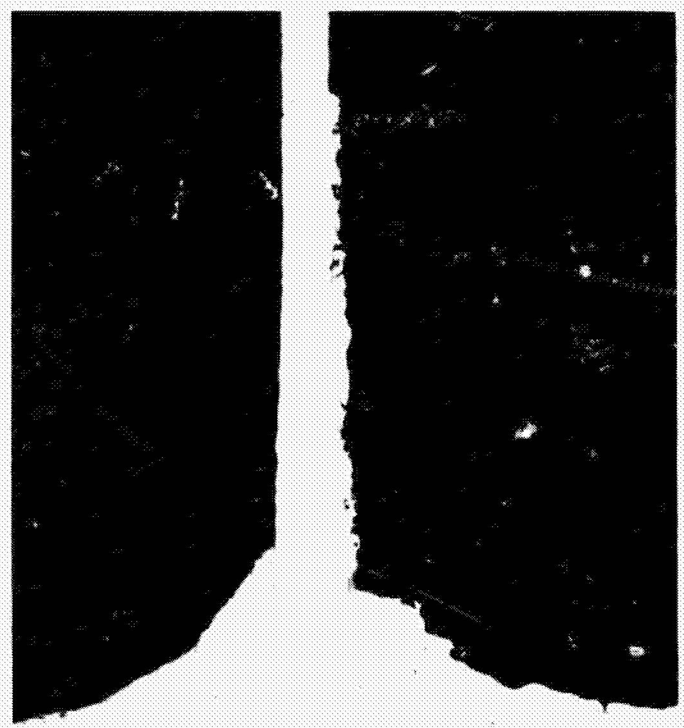


Figure 27. Laboratory Immersion Testing of PH14-8 Honeycomb, Twenty- Four-Hour-Exposure Specimen, Face-to-Face Core Braze, Magnification 200X

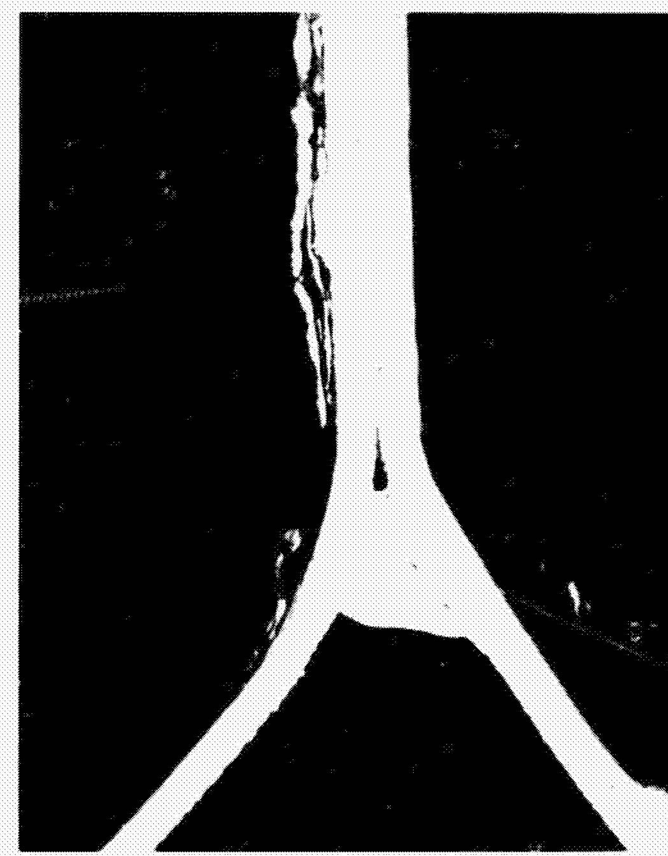


Figure 28. Laboratory Immersion Testing of PH14-8 Honeycomb, Twenty- Four-Hour-Exposure Specimen, Section at Meniscus, Magnification 200X

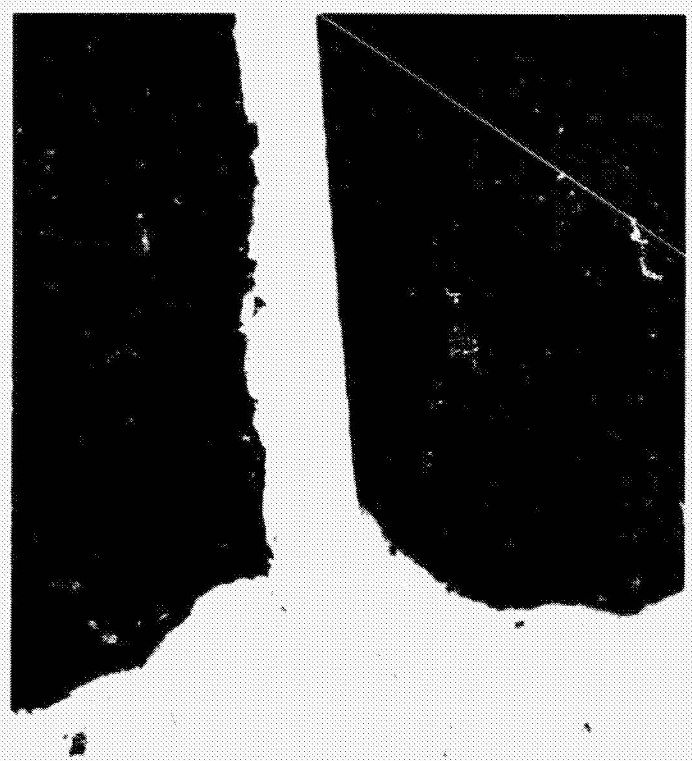


Figure 29. Laboratory Immersion Testing of PH14-8 Honeycomb, Seventy-Hour-Exposure Specimen, Face-to-Core Braza, Magnification 200X

Figure 30. Laboratory Immersion Testing of PH14-8 Honeycomb, Seventy- Hour-Exposure Specimen, Section at Meniscus, Magnification 200X

Data obtained from Armco stated that notable pitting of PH14-8 MO material was evident after immersion in ocean water for four days. It was also determined that, once initiated, the corrosion is essentially self-perpetuating.

Therefore it is concluded that the critical immersion time is between 24 and 74. Also, based on this limited test data, the maximum allowable exposure time prior to neutralization must be limited to 24 hours. Heat shield assemblies exposed to sea water or residual chloride deposits longer than 24 hours cannot be used for structural applications.

4. Residual chlorides when exposed to high relative humidity are equally, or more corrosive, to PH14-8MO material than full submergence in sea water. Therefore, reuse of the heat shield after any exposure to sea water is contingent upon a satisfactory flushing and neutralization system.

PREFLIGHT CORROSION INHIBITION TREATMENTS

The principal concern of this portion of the report is the determination of methods to minimize the corrosion effects of sea-water exposure on the CM structure and systems materials by use of minimum preflight precautions. There would be sufficient lead time, with most of the Block II vehicles, to permit institution of simple preflight treatments which will be very effective in minimizing potential corrosion damage if an early NASA approval were received.

Specific sealing requirements and recommended modifications of the brazed heat shields, to minimize corrosion and implement neutralization of the corroding chlorides, are described with the proposed postflight neutralization studies.

Corrosion of the external surface of the crew compartment and systems materials can best be minimized by the following procedure.

1. Apply RTV sealers on all exposed bare aluminum structure and conformal coatings to all electrical terminal attachments with particular care.

2. Mist-spray all adhesive-primed structure, plumbing, and storage equipment, and electrical terminals with an approved silicone material in an air-dry suspension medium before installation of the heat shield. The same material may be applied to the inner surface of the heat shield before installation of the insulation.

As an alternate method, an approved coating could be applied to the metal details at an earlier stage of fabrication, and the final mist-spray application would then be localized.

3. Serious consideration should also be given to development of a method to seal the edges of the insulation blankets thus preventing sea water saturation and the subsequent difficulty of eliminating the chlorides.

POSTFLIGHT NEUTRALIZATION OF CORROSION MEDIA, LABORATORY STUDY

Purpose

The external surfaces of the heat shields may be adequately neutralized by the postrecovery flushing procedures described later in this report. Development of a feasible means to remove the sea water from within the cells, neutralization of the active chlorides, and establishment of a non-destructive inspection technique were the principal objectives of a preliminary investigation undertaken in support of this RCM study.

Specimen Preparation

It was ascertained that a direct correlation between the corrosive attack on the aft heat shields of S/C's 009 and 011 and the Block II vehicles was impossible because the core in the early heat shields was not perforated. A brazed heat shield panel, which had perforated core, was therefore obtained for this investigation. The core configuration was identical to that of the production heat shields. The depth of the core in the test panel, however, was 1/2-inch thick instead of 2-1/2 inches. None of the later material was available. The test panel was approximately 15 inches in diameter and a hole was drilled through the center of one facing. A tubular part, for attachment of a hose, was centered on the hole and welded to the drilled face. The edge of the panel remained unsealed to permit uniform flow of the test fluids through the exposed core.

Test Procedures

Panel flow and purge tests were conducted with the test setup illustrated in Figure 31.

1. Flow tests were conducted to determine the minimum static head pressure at which sea water would fill all the cells, and the minimum positive and/or negative pressure required to maintain a constant flow of purging fluids through the panel. It was determined that the best uniformity of fluid circulation was obtained

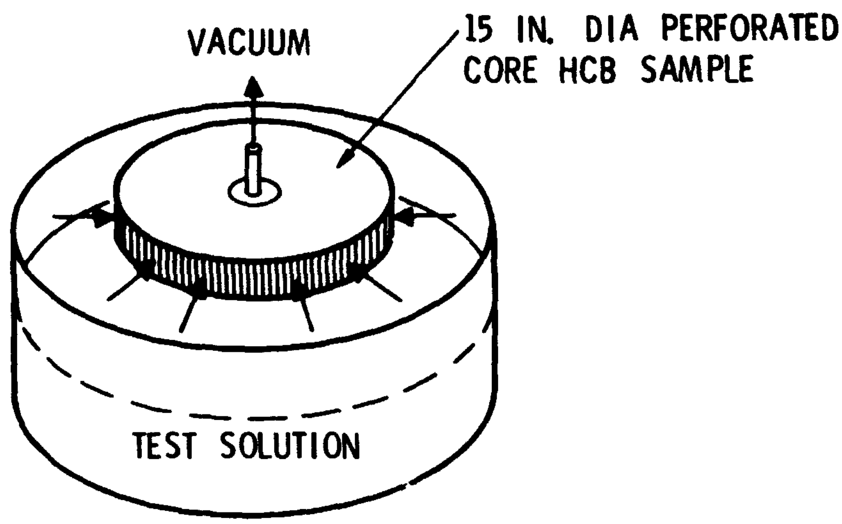


Figure 31. Flow and Purge Test Equipment

by submerging the panel in the purging fluid while maintaining a negative pressure on a port attached to the center of the panel.

2. The purge and neutralization tests were conducted with four purging solutions in the following sequence:

   First, de-ionized water; second, isopropyl alcohol; third, Freon-TF liquid; and finally liquid nitrogen. The panel was then dried by baking for approximately one hour at 160 to 200 F under a partial vacuum. It was ascertained that several volumes of purging fluid were required for each state of neutralization and drying. Subsequent to the final drying, a segment of the panel face was removed to visually inspect and chemically test for uniformity of purge and possible residual chlorides.

Results and Recommendations

1. Both positive and negative flow pressures were determined to be the minimum acceptable when the liquid flowed continuously without air bubbles. The minimum positive pressure was determined to be 1 psig. The minimum negative pressure was three inches of Hg. It was also determined that all cells of the core would fill when the panel was submerged under six inches of

sea water. It was therefore concluded that the surface tension of the sea water would not prevent the core cells from being filled, and little or no pressure head was required to completely fill the core of the heat shield.

2. It required six changes of demineralized H_2O to reduce the chloride content of the effluent of the test panel to less than 10 parts-per-million of H_2O. Six changes of each subsequent purging fluid were also used. Upon examination of the stripped panel, it was ascertained that no chlorides were entrapped and the feasibility of the process was substantiated. However, before this process can be applied for a proposed refurbishable heat shield it must be proven on a representative full-sized (or quarter-segment) heat shield and modified for shipboard operations. Subsequent to purging and drying, the panel must be stored in a controlled environment. The high-humidity, saline-containing mist environment would induce active new corrosion.

MODIFICATION OF FLIGHT HARDWARE

Aft Heat Shield

To adapt this purging process to a full-size heat shield, the assembly would require modification prior to making the initial flight and before mating the aft heat shield to the side heat shield in a final assembly. The minimum modification for purging would consist of the following: A minimum of one vacuum-port attachment must be provided at the centroid of field areas of the core in each segment of the brazed heat shield. A single port should be provided at the center of the assembly to permit introduction of the purging fluids and equal distribution around the edges of the perforated core. Some peripheral seal of the edge members may be required to minimize vacuum leaks at the spotwelded- and riveted-edge members. The vacuum ports and center port may be plumbed to permit ease of access on board ship, even in a partially disassembled condition, thus minimizing the time the core is exposed to sea water. The added weight may be compensated for by trimming the ballast sheets located on the inner surface of the aft heat shield.

These modifications may be varied to permit trepanning either face of the heat shield, in strategic locations, and attaching valve-stem type ports for the flushing operation. The valve stems would be held in place by standard vacuum-bag techniques. The need of permanent plumbing would thus be negated.

Crew Compartment Side Heat Shield

The current configuration of the side heat shields has been examined. The core material in these assemblies is also perforated. The edge members

have had 3/32-inch diameter holes drilled through, to the core, to permit circulation of inert gases around the periphery of the individual panels as part of the welding fabrication procedures.

The lower portion of the heat shield is submerged in sea water after splash down, and in its present configuration the core cells in that portion may become filled with sea water. Therefore, the panels should be sealed prior to initial flight by plug welding the holes and/or by adequate use of RTV type sealing compounds. The assembly should be carefully sealed at all points which might permit seepage of sea water into the structure. A procedure should also be provided to vacuum-proof test the assembly and prove the adequacy of the seal. The proof test should also be adaptable to postflight inspections. The adequacy of the sealing method should also be tested on a flight vehicle prior to the Block II vehicles.

PROCEDURE REQUIRING NO FLIGHT HARDWARE MODIFICATION

A relatively simple purging procedure not requiring flight hardware modification, but instead the removal of the aft heat shield aboard ship, is outlined below.

After recovery, and prior to 24 hours after splashdown, the aft heat shield is separated from the spacecraft and the thermal insulation blankets are removed. After this, the accessible regions of the spacecraft, which were exposed to the sea water, are thoroughly flushed with deionized water.

The residue water is then removed from the heat shield by means of the vacuum system used for purging. This obviates the necessity for tipping the heat shield. The four purge fittings are then installed in each of the honeycomb quadrants in the prescribed locations (at 0.59 radius from the center). The manifold connecting these fittings to the vacuum system is then installed (Figure 32), and the flushing begins. Purging fluid, consisting of 8.5 volumes, is poured into the aft heat shield and the vacuum system activated. A tarpaulin cover should be placed over the heat shield when the volatile fluids (isopropyl alcohol and liquid freon) are being used. In this manner the aft heat shield honeycomb is sequentially flushed with deionized water (chloride getter), isopropyl alcohol (water getter), and finally liquid freon (very volatile alcohol getter). Approximately 40 gallons (8.5 volumes) of each flushing fluid is used.

POSTFLIGHT STRUCTURE AND SYSTEMS MATERIAL EVALUATION

Evaluation of the crew compartment and the heat shield structures for additional flight missions is contingent upon thorough postflight inspection for corrosion and latent structural damage. The reliability of the support

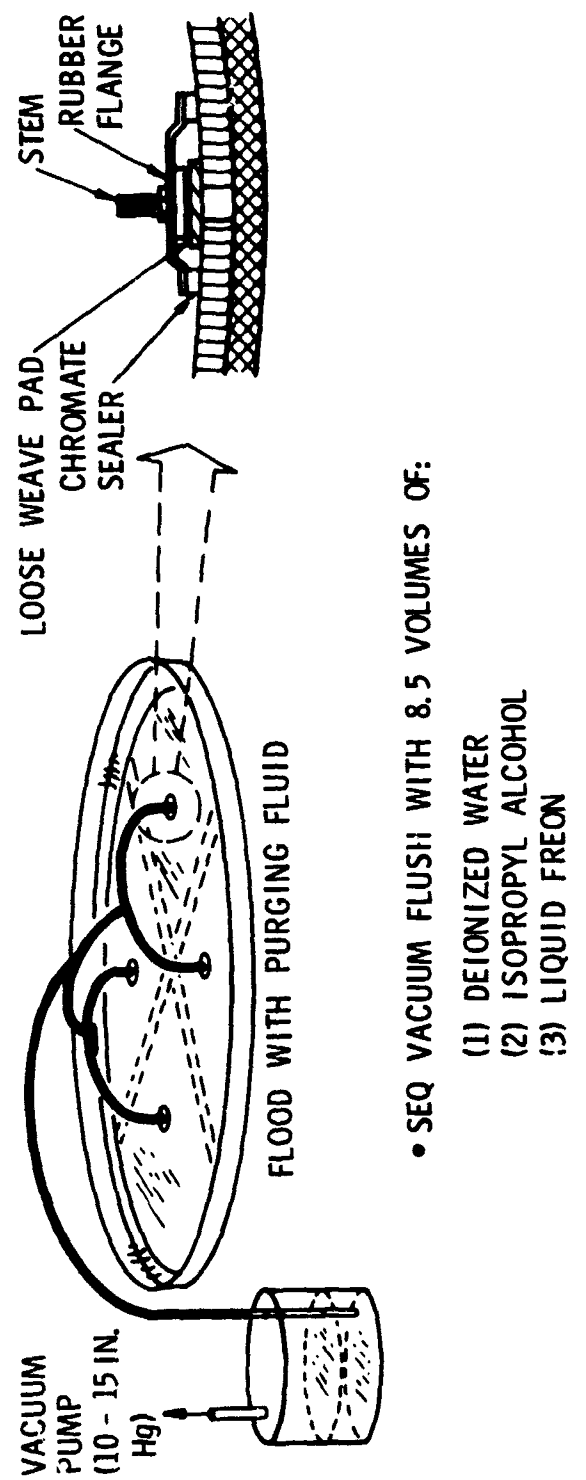


Figure 32. Minimum Plumbing Layout Heat Shield Purge

systems is also contingent upon detection and elimination of potential corrosive degradation. In many instances local repairs may be made to refurbish the structure or system for reliable service.

Postflight Inspection for Corrosive Damage

Latent damage to the structure and systems materials, attributable to immersion in sea water, cannot be determined by visual inspection alone. Local non-destructive test methods may be used to detect the presence of chloride deposits. Suspect parts may then be evaluated by proof- and local-limited destructive tests to supplement the visual analysis. These tests are not complicated, and subject items are not too extensive in number.

Crew Compartment

The external surfaces of the aluminum facings have been coated with a modified epoxy adhesive primer. Corrosion of the aluminum materials will therefore be localized in areas where the prime coat is damaged and where hairline scratches or voids occur in the primer. The salt crystals and residue must be flushed from the surface of the primer, and a fluorescent or equivalent spot check must be made to ascertain if aluminum material has been exposed through the primer and thus subjected to the corrosive sea water. If a positive exposure is indicated, the primer must be locally stripped and the extent of the corrosion damage evaluated. The complete external surface of the crew compartment must be thus inspected.

If corrosion has started in the faying surfaces between edge members and facings or between facings and doublers, the suspect bond joint must be stripped, to remove the corrosion, and prepared for rebonding. If this repair is not made, the corrosion could perpetuate and permeate the entire joint and abutting structure.

Electrical Terminals

RTV type elastomers and conformal coatings have been applied to some of the electrical terminals subsequent to securing the electrical leads. However, the coating application has not completely encapsulated some of the terminals and corrosion is inevitable after sea-water immersion. The chemical spot-check method may be used to detect the presence of corrosive material. All suspect terminals should also be checked electrically.

Brazed Heat Shields

It is assumed that adequate provisions have been made to seal the side heat shield and modify the aft heat shield for [illegible]recovery flushing. The recommended revisions are described [illegible] the section of this report.

The following procedure shall be followed when conducting a postflight inspection and evaluation of the modified heat shields for repair and refurbishment.

1. Deionized water shall be pumped through the individual segments of the aft heat shield and a silver nitrate precipitation test shall be conducted to detect the presence of chlorides. If the sidewall has been immersed in sea water, it shall again be leak tested to determine the adequacy of the original panel sealing. A seal failure shall necessitate a precipitation test being run to detect the possible presence of chloride within the sandwich core. This will require postflight modification of the panel for attachment of purging parts.

2. X-ray inspect the steel honeycomb substructure to ascertain the presence of face-to-core bond voids, weld cracks, damage to upper and lower rings and vertical structure, and overall distortions.

Postflight Inspection for Structural Bond Damage

The bonded crew compartment structure has been designed to meet the stipulated parameters for lunar flight. Any areas of local destructive damage may readily be observed by visual inspection and necessary repair procedures instituted.

Latent damage to the bonded structure cannot, however, be readily ascertained by visual examination. Pertinent proof loading, nondestructive test techniques, and local destructive tests must be conducted to supplement the visual analysis of the qualified postflight inspection team.

Inspection of the crew compartment for structural damage by use of the production ultrasonic techniques, using immersion or squirted water for signal transmittal, should be avoided at all cost. Sealing of the complete compartment to prevent water seepage into the sandwich is impractical, if not impossible. Detection of water inside the sandwich is very difficult and the current methods are not adaptable to the completely bonded structure. Bond evaluation by ultrasonic methods using a small transducer and local coupling agents is impractical for the large areas requiring inspection. Many areas will also be inaccessible to the probe equipment. A combination of the following methods is therefore recommended.

1. Remove all unwanted shelving, compartments, and attaching clips before inspecting for cracks in bond lines and delaminations in face-to-core bonds. This must be done carefully to prevent damage to the sandwich.

2. Apply torque proof loads, as originally applied in proofing the bonds on the production command modules, to the channels, clips, and fittings which will be subjected to shear, tension and torque loads during the proposed missions. This proof loading should also be completed before inspecting for cracks and delaminations.

3. Check all channels, fittings, and attaching clips to determine presence of cracks in the bondlines around the edges of the secondarily bonded details. Visual inspection and use of feeler gages may be used to detect larger voids in conformance with NAA Specifications MA0606-006 and MA0606-014. The size of the voids must be checked against the production records of the spacecraft being refurbished. Any increase in void size shall be reason for removal and replacement of the secondary bonded detail.

   Many cracks may not be readily discernible by visual inspection, particularly in the areas made relatively inaccessible by attached equipment, plumbing, and electrical harness installations. These areas may be checked for cracks by a dye-penetrant procedure. Proper material selection shall be made to prevent damage to the bond or details. Areas indicating cracks or voids shall be subject to engineering review to determine the necessity of detail removal and replacement.

4. After all defective secondary bonded details are removed, the assembly shall be checked for face-to-core bond delaminations. The initial check should be made by regulated tapping, using a serrated rowel with an amplifier pickup attachment. The amplified signal may be fed through earphones or a speaker system. The edges of the serrated wheel shall be coated with a plastic material to prevent damage to the metallic faces. This system of detection has been used successfully by NAA for quality control of production bonding. Gross voids and suspect areas are readily detected.

   The suspect areas shall be further inspected by use of pulse-echo equipment using direct couplant for the transducer-to-bond coupling. Care shall be taken in selection of a couplant which will not contaminate the areas where subsequent bonding may be required. A fluorescent trace element may be added to the couplant which will permit detection of the couplant contaminated areas and simplify the cleaning procedures.

5. Voids detected by the above procedures shall be subject to engineering evaluation and proper repair procedures shall be instituted.

6. When large areas of the crew compartment require repairing, the face-to-core bond strength of the outer facing shall be proof tested by the established trepanned-tensile-test techniques. The number and location of the test specimens shall be determined by engineering review.

7. After all repair and rebonding is completed, the escape hatch and tunnel hatch shall be replaced and the crew compartment shall be pressure tested in conformance with the original structural requirements. Any leaks shall be traced and the suspect area delineated. Complete refurbishment of the hatch seals may be necessary.

Repairs (Bonded Structure)

In some instances, void repairs may be accomplished by paste or foam adhesive in conformance with NAA Specification MA0606-006. In other instances, the outer facing and delaminated core may be removed and replaced with new materials in conformance with established repair procedures (NAA S&ID PUB 543-G-18). Allowance shall be made for reduced core-to-face bond strength commensurate with bonding to precured adhesive residue on the inner face of the crew compartment. Complete removal of the old adhesive and adequate recleaning of the base metal is difficult to accomplish without inflicting additional damage to the surrounding core during the cleaning process. Some additional effort should be expended to refine the repair procedures to accommodate the epoxy-phenolic adhesive systems. The core splices may be adequately achieved with the approved epoxy-phenolic foam or paste materials.

Subsequent to completion of repairs and rebonding, the trimmed edges of the structure having bare aluminum exposed shall be covered with an RTV sealing compound. An additional corrosion-inhibiting coating may be applied over the entire structure to improve the corrosion resistance.

Postflight Evaluation of Brazed Heat Shields

Ablator Refurbishment

Localized repairs of the ablator may be accomplished at S&ID. Major repairs can be best accomplished at Avco where the assembly must go for resizing to the new aerodynamic contour. Minor localized gutting of the ablator may be repaired by removing the charred material and applying lightweight putty type ablation materials, such as Avco 5026-39, Dow Corning DC-325, or Emerson Electric's T500-111. Major repairs to the ablator can best be repaired by locally removing all ablator and core

..iforcement down to the steel sandwich face. New material may then be applied in the original sequence (i.e., bonding unfilled core to metal substrate and filling with Avco 5026-39), and sizing to the aerodynamic contour.

Crew Compartment Side Heat Shield

The salvaged crew compartment side heat shield assemblies from S/C 009 and 011 indicated no serious damage to the steel sandwich structure. It may therefore be expected that this structure, properly sealed before flight, would also be usable with possible minor repairs. This assumption is contingent upon satisfactory results from the postflight inspection for corrosion.

Aft Heat Shield

The aft heat shield will probably experience the greatest damage, due to impact. However, every effort should be made to salvage the structure, if at all possible. The toroidal section may be removed in segments and locally replaced. Damaged pie-shaped segments may be removed and new segments welded in place.

Every effort should be made to mate the same heat shields with the matching crew compartment, since indexing of the crew compartment-to-heat shield channels is not identical for all spacecraft.

PLASTIC BOOST PROTECTIVE COVER

The boost protective cover is jettisoned after take off and must therefore be replaced with a new assembly. Special note should be taken of the fact that the replacement assembly will probably require new fixtures and tools for fabrication. Each cover is custom tailored to match a specific command module. The RCM will be of a different contour than any of the Block II configurations. Allowance must also be made for fabrication of new core-forming fixtures by the core supplier. The core-forming techniques are critical and forming cannot be accomplished on a rework basis.

POSTRECOVERY OPERATION REQUIREMENTS

The current recovery procedures were established commensurate with the requirements of a single flight mission. The procedures, therefore, do not protect the heat shields or the structure and systems materials from the corrosive effect of sea water. The exposure is not limited to the time between splash down and recovery on board ship, but includes subsequent action of sea water entrapped in the heat shield and saturating the insulation, and the high-relative humidity and high-chloride content environment during transport to dockside.

If postrecovery procedure modifications equivalent to the following are introduced, much of the structure and the systems material, which is currently subject to corrosion and subsequent scrappage, may be usable for later flights.

1. The escape hatch of the inner compartment should be relocked, if possible, as soon as the astronauts are out of the CM. Internal corrosive conditions would be further minimized if desiccants were placed inside the cabin.

2. Subsequent to recovery on the deck of the carrier, the aft heat shield should be removed, to permit removal of the saturated thermal pads and facilitate flushing of the space between the side and aft heat shields and the inner compartment. Fresh water, Preferably demineralized, should be used. A spray rinse would not be effective because of the type and amount of insulation within the space.

3. The aft heat shield should be thoroughly flushed and neutralized in accordance with an approved procedure commensurate with that evaluated in the preliminary corrosion studies and included in this report.

4. The aft heat shield, subsequent to flushing and drying, or a substitute fairing shall be attached to the aft end of the CM to prevent damage to the thin facings on the crew compartment aft bulkhead during transport.

5. Corrosion would be further minimized if the complete compartment were dryed at 160-200 F (in a simple portable oven) for 3 to 6 hours, subsequent to flushing and neutralizing, and placed within a controlled-humidity tent during trans-shipment.

EXPLORATORY TEST REQUIREMENTS, CLASSIFICATION TESTS

There are two basic classes of exploratory testing which must be considered to establish the integrity of the materials used in the basic CM structure, and those of constituent individual systems used to achieve the mission objective. One type of test determines the effect the initial flight has had on the basic properties of the materials. This type of test needs to be conducted only once on a piece of representative or actual flight hardware which has been subjected to all the parametric variables of the initial flight mission. The information thus obtained can be applied for all future hardware from identical flight missions. The second type of test must be repeated for every flight vehicle to assure the reliability of later flights.

Single Term Tests

Those tests required to establish a process for purging and neutralizing the aft heat shield have been described in a previous portion of this section.

There are several other single-term tests which should be conducted to evaluated systems and components for potential multiple-service operation or for single-term use in the proposed RCM environment.

Reaction Control System Engines

The following general tests should be conducted to qualify the SM-RCS for proposed RCM Laboratory service requirements.

1. Qualification of the entire subsystem to the extent necessary to assure satisfactory operation under the conditions of exterior mounting to which it will be exposed. Qualification by similarity may be possible.

2. Qualification of pressure vessels for the additional mission time to which they will be exposed under pressure.

Additional development verification of the RCS engines is not recommended at this time. The RCM laboratory attitude control requirements in terms of total burn time and cycles have not been defined. If it is determined that these requirements exceed the capabilities of the current engines, a determination regarding additional testing should be made at that time.

Command Module RCS Pressure Vessels

Postflight analysis of the RCS fuel tanks of S/C 011 revealed the presence of internal corrosion and excessive contaminant particles. At least one more postflight system tank should be checked out before receiving approval for the RCM mission. In addition, pressure vessels similar to those which will be reused should be qualified for the cumulative total of the original mission plus the 14-day RCM spacecraft mission.

Cold Plates

1. The cold plates on CM's 009 and 011 were not of brazed construction. A flight article of the brazed configuration should be sectioned and examined before being approved for the RCM mission.

2. Note: NAA has experienced corrosion with cold plates in which contaminated ethylene glycol had been circulated. NAA also experienced corrosion of cold plates from ethylene glycol residue in water. Improved methods of flushing must be provided.

Plumbing Lines:

It has been concluded that replacement constitutes the optimum renovation approach in regards to plumbing lines.

Windows

The optical characteristics (light transmission and UV absorption) will have to be determined for the windows (i.e., the window panes of the outer structure and inner structure) of the recovered Apollo command modules selected for RCM usage.

Visual examination of S/C 009 and 001 windows performed at S&ID disclosed deposits of residues on the outer panes. The windows of the S/C's 009 and 011 were forwarded to NASA-MSC (at the request of NASA) for comprehensive study purposes.

Preliminary evaluation of the S/C 011 by NAA revealed that the residue deposits were restricted to the window pane of the outer structure (Figure 33) and indicated that the residue may be removed by washing. However, the postflight evaluation at S&ID has not been completed. The evaluation should be completed on a subsequent flight.

Thermal Control Coatings

The RCM will require new radiator locations. The new thermal control coating will likely be required based upon a new solar absorptance/emittance ratio (α_s/ϵ) requirement. The white coating (Type Z-93, zinc oxide pigment in potassium silicate binder) used on the Apollo radiators will be used to the greatest extent possible, probably in conjunction with another coating also known to be stable to space environment. The thermal-control coatings selected will have to be subjected to development verification and to component-verification qualification tests.

Latching Mechanisms

Tests should be planned for those subassemblies, such as the latching mechanisms for the crew hatch, which are non-lubricated and where the RCM missions will require extended use for these assemblies.

Latch type mechanisms such as those used for the crew hatch might have to be re-designed for the RCM missions.

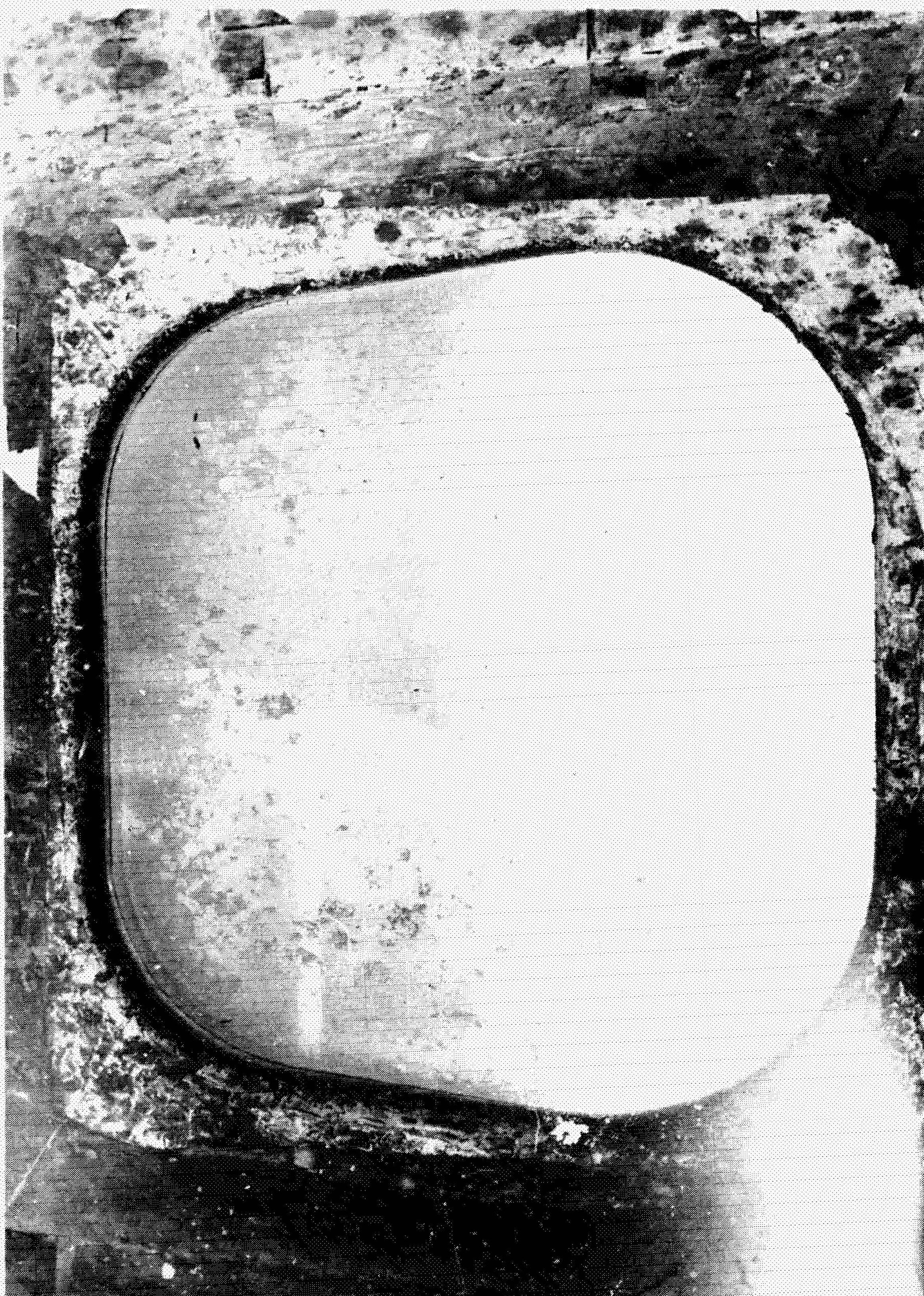


Figure 33. Spacecraft 011 Window

Vacuum Sublimating Materials

Considerable data reflecting environmental laboratory testing of dry film and viscous lubricants, non-metallic materials, and any materials in general which are subject to significant sublimation in vacuum, are available and are supplemented by actual satellite experience. Where data for materials, lubricants, etc., similar to those used on Apollo are available, and indicate the suitability of these materials for vacuum exposure compatible with the RCM mission length, qualification by similarity would be employed to reduce the amount of required testing.

Dry Film Lubricant

Tests should be planned to determine the adequacy of the Apollo designs which employ dry-film lubricants for the extended RCM applications. Limited sliding-friction tests should be conducted which represent the material combinations and contemplated RCM applications.

Solid dry-film lubricants are used for general Apollo applications where sliding friction is involved. The extended performance of these lubricated surfaces, especially after long time exposure to space environment, is not known.

Viscous Lubricants

The viscous lubricants currently used for Apollo applications must be re-evaluated on the basis of the extended mission parameters. Current approval cannot be construed as unlimited acceptance for the alternate missions.

Nonmetallic Material Environment Testing

The materials currently used within the Apollo CM have been approved on the basis of the current crew compartment environment and a maximum flight exposure of 14 days.

The proposed environment is changed and many of the materials may prove to be time dependent. Therefore, the non-metallic materials must be re-evaluated to ascertain conformance to the flammability and toxicity requirements as noted in the Apollo Material and Producibility Bulletin Number 8. The tests shall be conducted in accordance with the governing NAA specifications (MA0115-008 and MA0115-009) with the exception that the test chamber requirements shall be modified to represent those of the extended mission, and the exposure time shall be extended accordingly.

Threaded Mechanical Parts

Special endurance tests for fasteners and threaded mechanical parts (to be subsequently designated) in line with their planned RCM extended use in space environmental conditions (vacuum, temperature) should be planned.

Apollo contains a small number of fasteners and threaded components largely confined to linkage type mechanisms associated with controls and which were designed for short time, limited usage. It is foreseen that the RCM will impose longer loading (stressed) times, extended cyclic loads, as well as the reuse of many of these components.

Pyrotechnical Devices

There are no requirements at present which involve vacuum exposure of pyrotechnics for longer than 14 days prior to activation; therefore, no additional qualification will be required. In the event that future missions entail separation or other pyrotechnic-activated functions after a longer exposure to the space environment, additional qualification may then be necessary.

Multiple Term Tests

Those tests required for corrosive and structure evaluation of the heat shields and crew compartments of all postflight vehicles have been described in a preceding section (Postflight Structure and Systems Material Evaluation).

The following tests, although of acceptance test stature, are mandatory to be assured of reliable material properties. Special note should be taken of the precautions which must be observed to be assured of requalification. The systems of particular importance are:

Cold Plates

The following tests should be planned for the cold plates of all recovered Apollo spacecraft:

1. Re-acceptance test procedures for command modules being considered for refurbishment should include analyses of test samples of the ethylene glycol solutions removed from the cold plate systems.

Cold plate systems whose solutions contain corrosive products or excessive quantities of water should not be considered for refurbishment.

Note: The water glycol fluid should not be drained unless the fluid can be completely removed from the cold plate system. The full water-glycol content should be retained and maintained if the water-glycol solution can not be completely removed. The fluid when maintained, should be periodically circulated in accordance with a planned schedule.

Reaction Control System Pressure Vessels

The following general tests should be planned for all recovered reaction control system tanks selected for reuse in an RCM.

1. A re-acceptance test procedure should be created and implemented for the RCS fuel, oxidizer, and pressurant tanks. Re-acceptance testing of the propellant tanks should consist of at least

   a. Visual inspections (disassembled)

   b. Dimensional inspections (disassembled)

   c. Proof test (without bladder and the other internal components)

   d. Leakage test (without bladder and the other internal components)

   e. Proof-cyclic tests (with bladder installed)

   f. Leakage tests (with bladder installed)

   The external tank surfaces and supplementary plumbing lines will have to be sprayed with fresh water to fully remove salt residues and then dried with isopropyl alcohol as soon after recovery from the original Apollo flight as is possible. In addition, proper flushing, evacuating, and neutralizing procedures will have to be implemented to the RCS, itself, and to the individual tanks.

IV. HEAT SHIELD

Figure 34 shows the three components of the heat shield. The forward heat shield, which is ejected prior to reentry, is not recovered and, therefore, must be replaced. As stated earlier, the crew-compartment heat shield suffers only minor degradation, from reentry plasma errosion and heating. However, the RCS propellant dump-burn prior to splashdown creates severe, but localized, charring (Figures 35 and 36). The degradation presents no serious renovation problem, since, at worst, the entire ablation material can be replaced in these areas without undue difficulty. The aft heat shield is degraded to a degree dependent on the reentry trajectory and, since it bears the brunt of the reentry stresses, is considered the critical heat shield component in this study.

A cursory analysis was conducted of the ablator performance of the aft heat. The analytic model developed in the Apollo program was used, and the results correlated to S/C 011 data to within 10 percent. Results of the analysis will be discussed in the following portion of this section, having been summarized in Section I.

ANALYSIS

The cursory analysis consisted of determining the thermal and physical response at two locations on the heat shield (the stagnation and leeward points) for two trajectories; both have heat loads equal to or greater than those for earth-orbital entry. A 28.5 kfps entry (S/C 011 nominal) and a maximum-heat-load-lunar-return trajectory were used, and nominal rather than design heating rates were used. Block II heat-shield thicknesses were used in analyzing the ablator performance, and the S/C 011 Block I thicknesses were used to ascertain nominal S/C 011 heat-shield temperatures.

The RCM spacecraft ablator-thickness requirements were established (sized) for the 28.5 kfps entry by determining the ablator thickness required to protect the bond line to a maximum temperature of 600 F before or at time of earth impact.

Results of the analysis are summarized in Figure 37. The NAA charring ablator program APD 113 was used for this analysis.

Figure 37 also shows the relative position on the CM of the locations considered. The one-dimensional model used for analysis is also shown for reference.

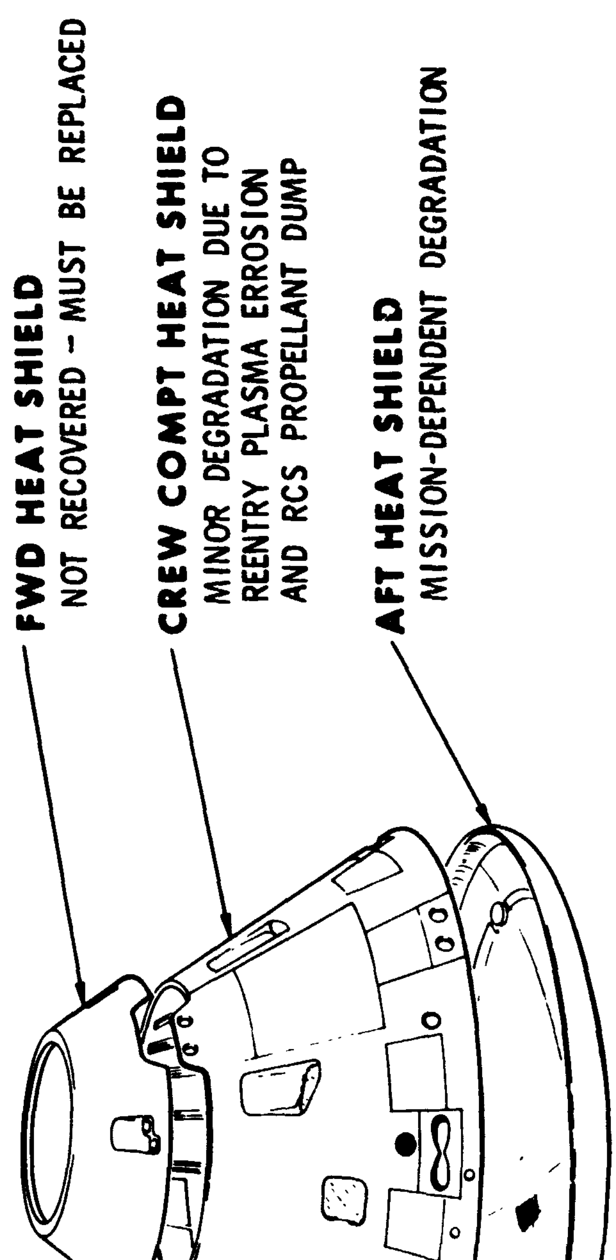


Figure 34. Block II Heat Shield

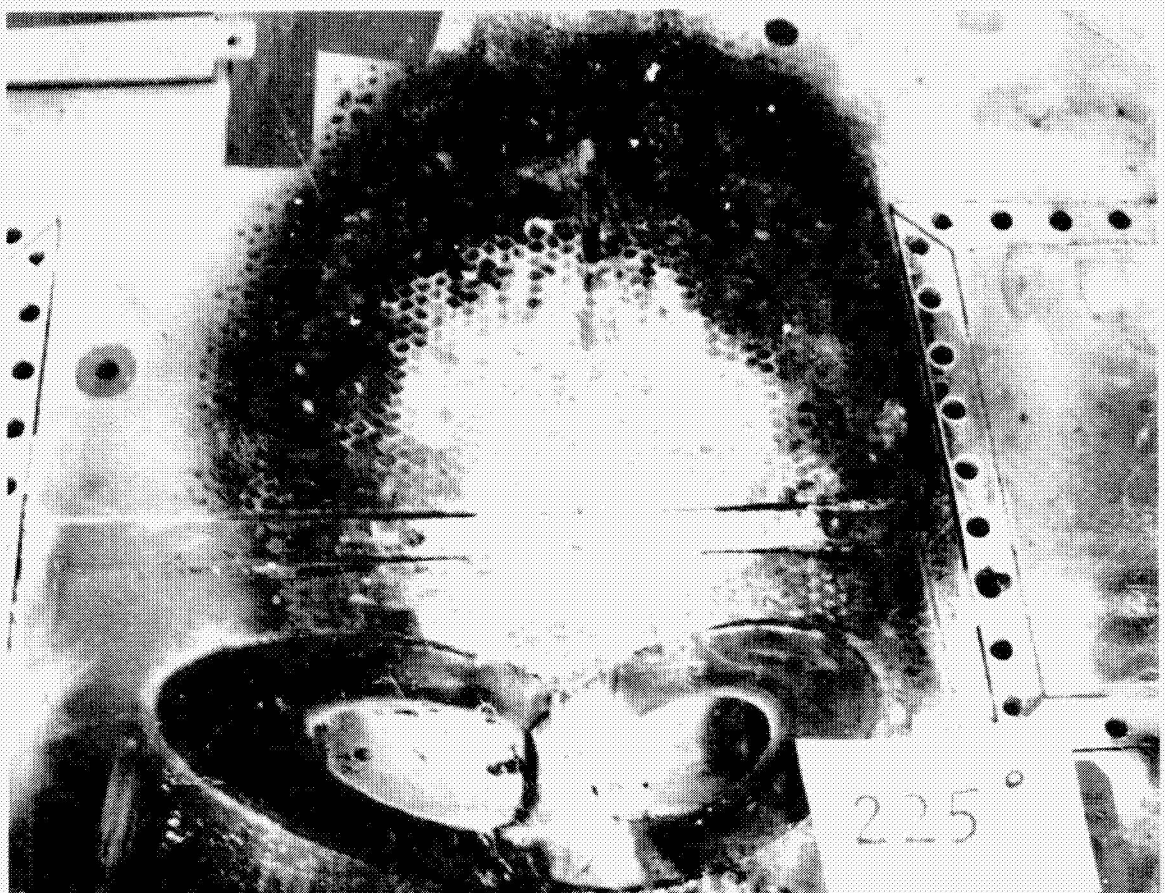
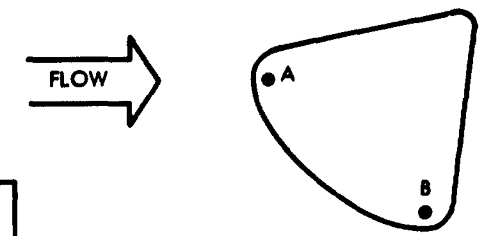
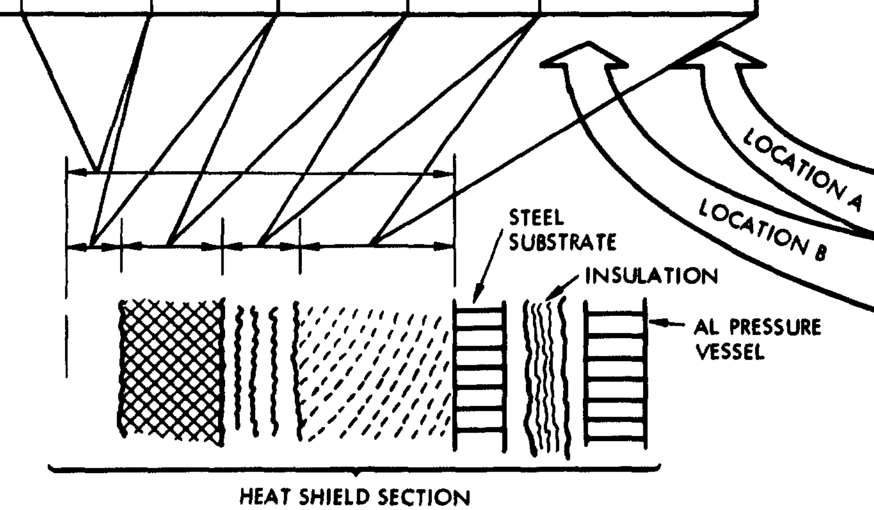


Figure 35. Spacecraft 011, Crew Compartment Heat-Shield Charring in Vicinity of RCS Roll Thruster, 225 Degrees

Figure 36. Spacecraft 011, Crew Compartment Heat-Shield Charring Vicinity of RCS Roll Thruster, 305 Degrees



APOLLO MISSION (NOMINAL TRAJECTORY)	LOCATION	INITIAL ABLATOR THICKNESS	SURFACE RECESSION	FULLY PYROLYZED	REACTION ZONE	REMAINING VIRGIN MATERIAL
ORBITAL	A	1.93	0.255	0.137	0.474	1.06
	B	1.53	0.114	0.114	0.461	0.84
LUNAR	A	1.93	0.386	0.143	0.532	0.87
	B	1.53	0.172	0.124	0.523	0.71



MINIMUM THICKNESS FOR RCM SC
0.94
0.80

$T_{BL} < 600$ F

ZONE CRITERIA:

CHAR- $\rho < 0.5 \rho_{V.M.}$

REACTION ZONE- $T > 600°F$

Figure 37. Aft Heat Shield Analysis

Figures 38 through 43 depict the bond-line temperature histories for the cases considered. Figures 44 and 45 present the aluminum pressure-vessel temperature history for the 28.5 kfps entry trajectory, with the ablator sized for this trajectory. The pressure-vessel temperature does not rise significantly in the cases where SC 011 final-machined ablator thicknesses and Block II thicknesses were considered.

By comparison (in Figure 37) of the virgin material remaining after flight with the sized thickness determined for the 28.5 kfps entry, it can be concluded that vehicles subjected to entry from earth orbit are potentially refurbishable from a thermal consideration. If this study were extended to other types of lunar-return entry trajectories such as maximum-heating rate trajectories, as opposed to the maximum-heating load trajectory that was considered, it is determined that some Block II vehicles may be refurbishable, based on individual mission considerations.

It should be stressed that this analysis and its results are preliminary. Areas of perturbed flow remain to be considered.

PRELIMINARY HEAT SHIELD RENOVATION ASSESSMENT BY AVCO

The appendix of this volume contains the Avco report of the preliminary study of heat shield renovation. It was conducted by Avco in support of the study, and categorized ablator renovation into three types:

Class A. Direct reuse of the ablator without any operation on the ablator material

Class B. Char removed and surface resealed

Class C. All the ablator material removed down to the substrate, and new ablator material installed and sealed

Class C renovation is technically feasible since it constitutes an Apollo Class 3 repair. Techniques and facilities exist, and the effectiveness of this approach has been proved in flight. The effectiveness and feasibility of Class A and B renovation has not been demonstrated and would require analysis of future Apollo heat shield char distribution, assessment of the thermostructural aspects of reusing ablator material, determination of aerothermodynamic performance of an ablator that has been once used, and the generation of production techniques to determine ablator status and salt water content and to preserve delamination. At the very worst, using Class C techniques, the renovated heat shield would cost approximately 65 percent the price of a new heat shield. If Class A and B techniques prove feasible, even greater savings will result from employing those

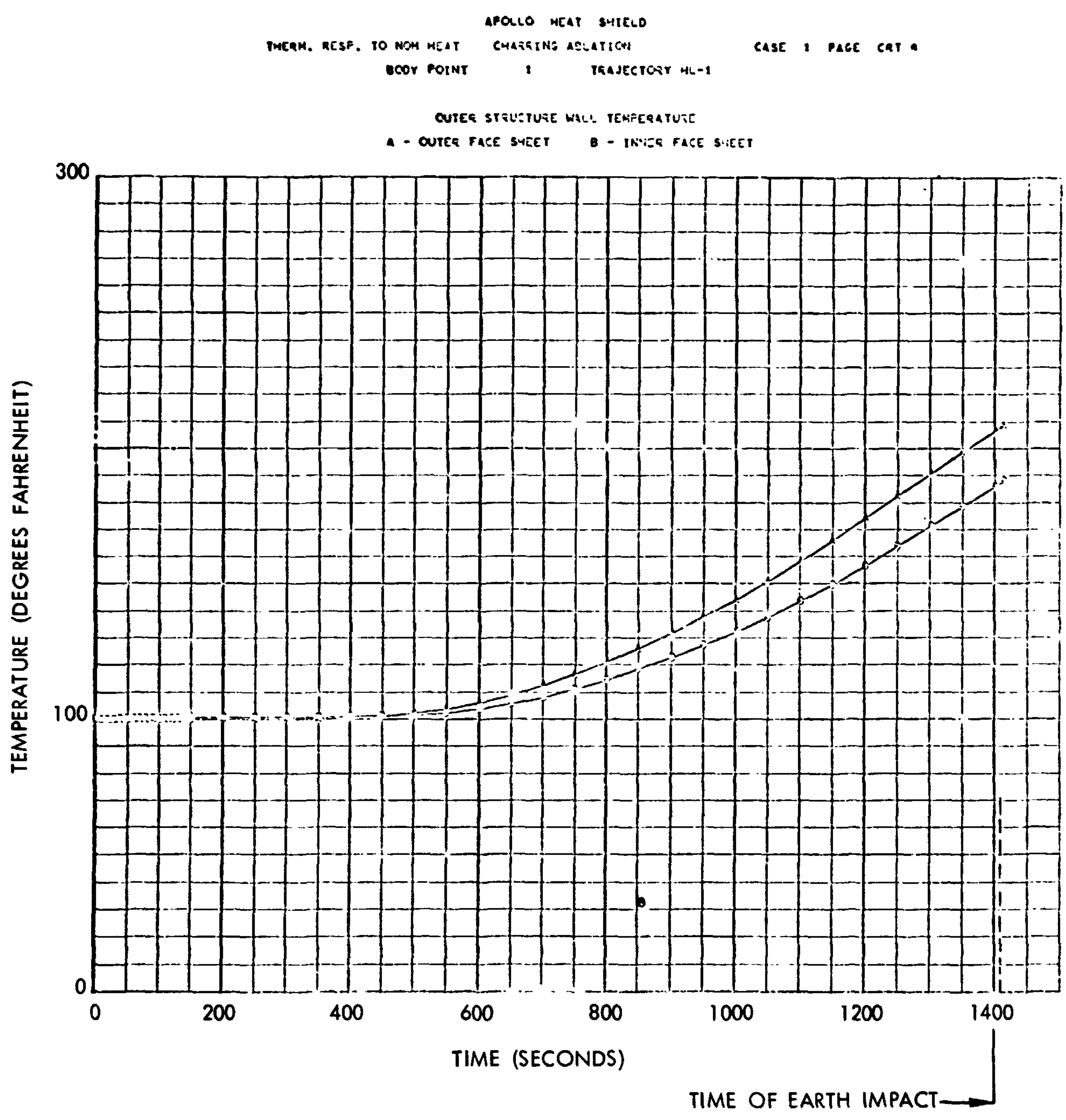


Figure 38. Bondline Temperature History, Location 200, Block II Thickness, HL-1, Nominal Heating

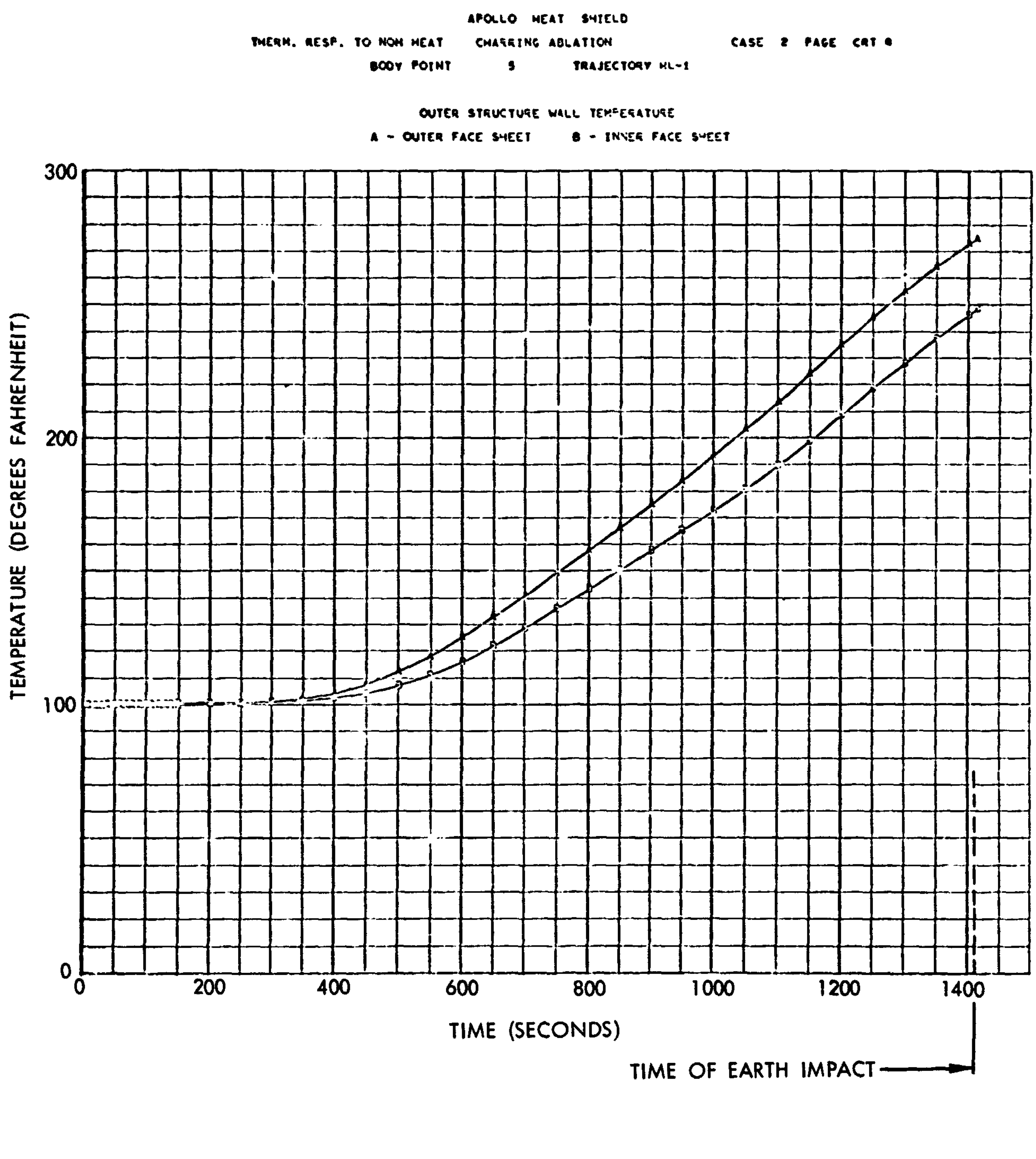


Figure 39. Bondline Temperature History, Location 210, Block II Thickness, HL-1, Nominal Heating

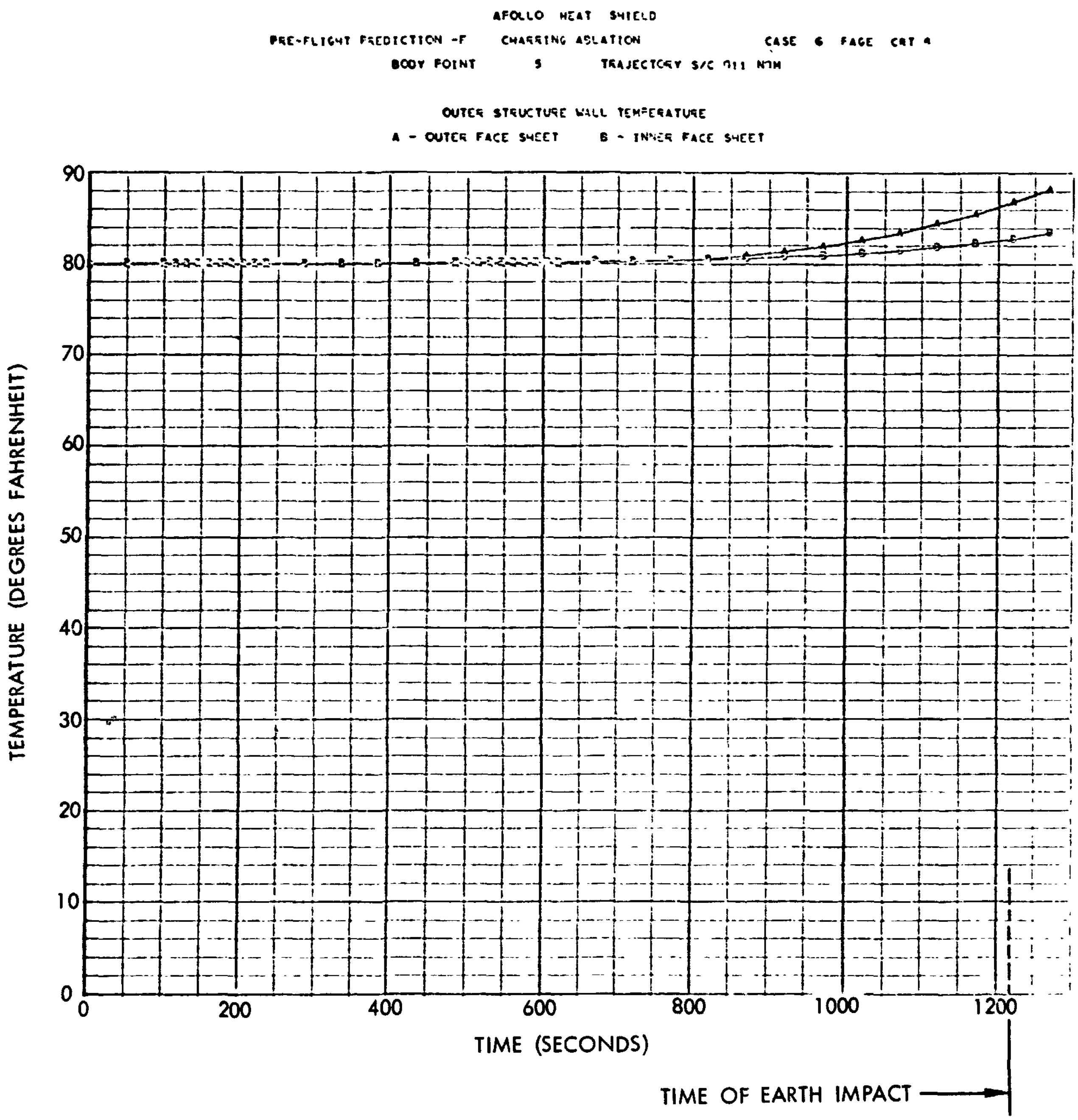


Figure 40. Bondline Temperature History, Location 200, S/C 011 Final Machined Thickness, 28,500-FPS Entry Trajectory

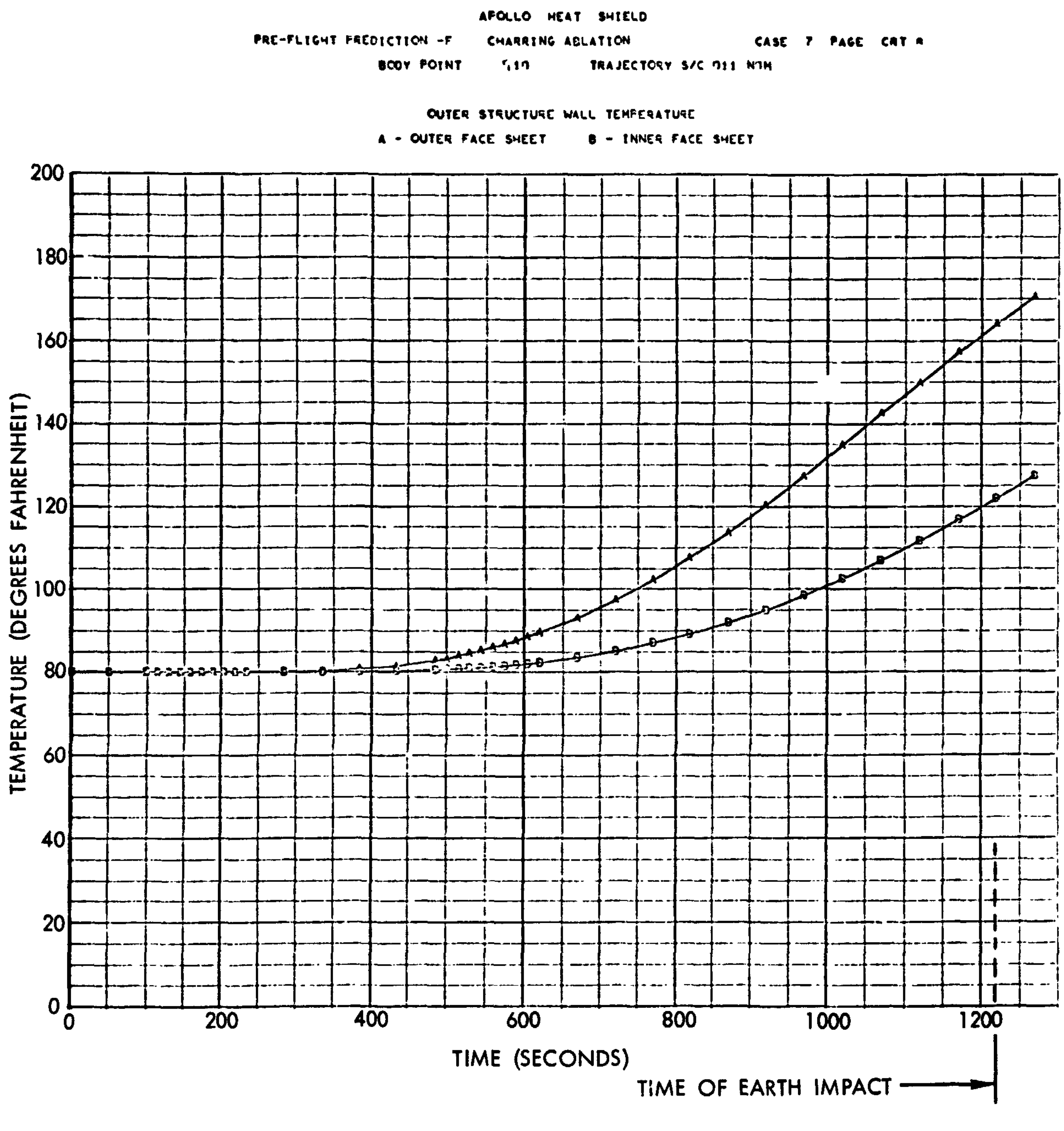


Figure 41. Bondline Temperature History, Location 210, Spacecraft 011 Final Machined Thickness, 28,500-FPS Entry Trajectory

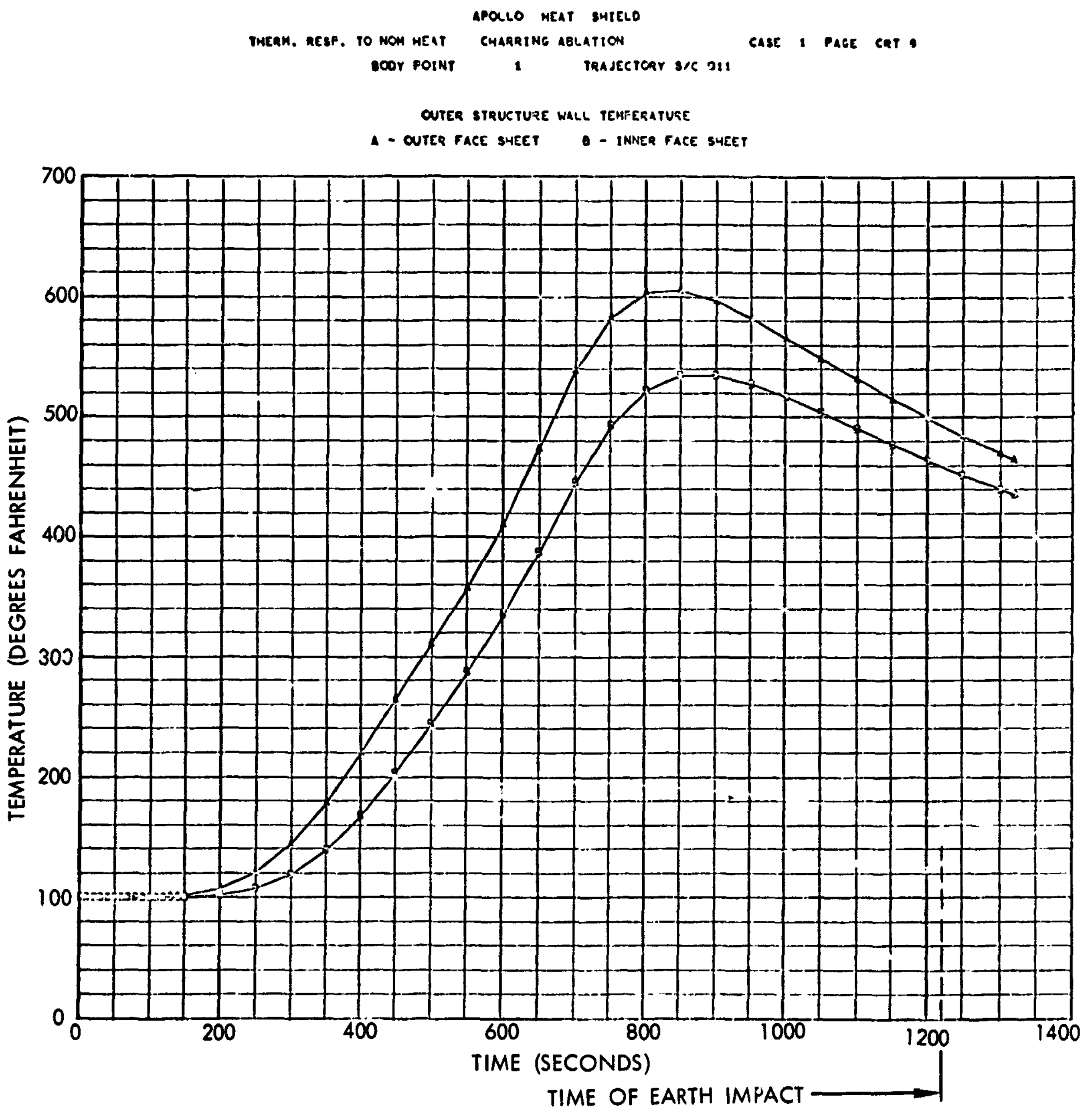


Figure 42. Bondline Temperature History, Location 200, Ablator Sized for, and Subjected to, 28,500-FPS

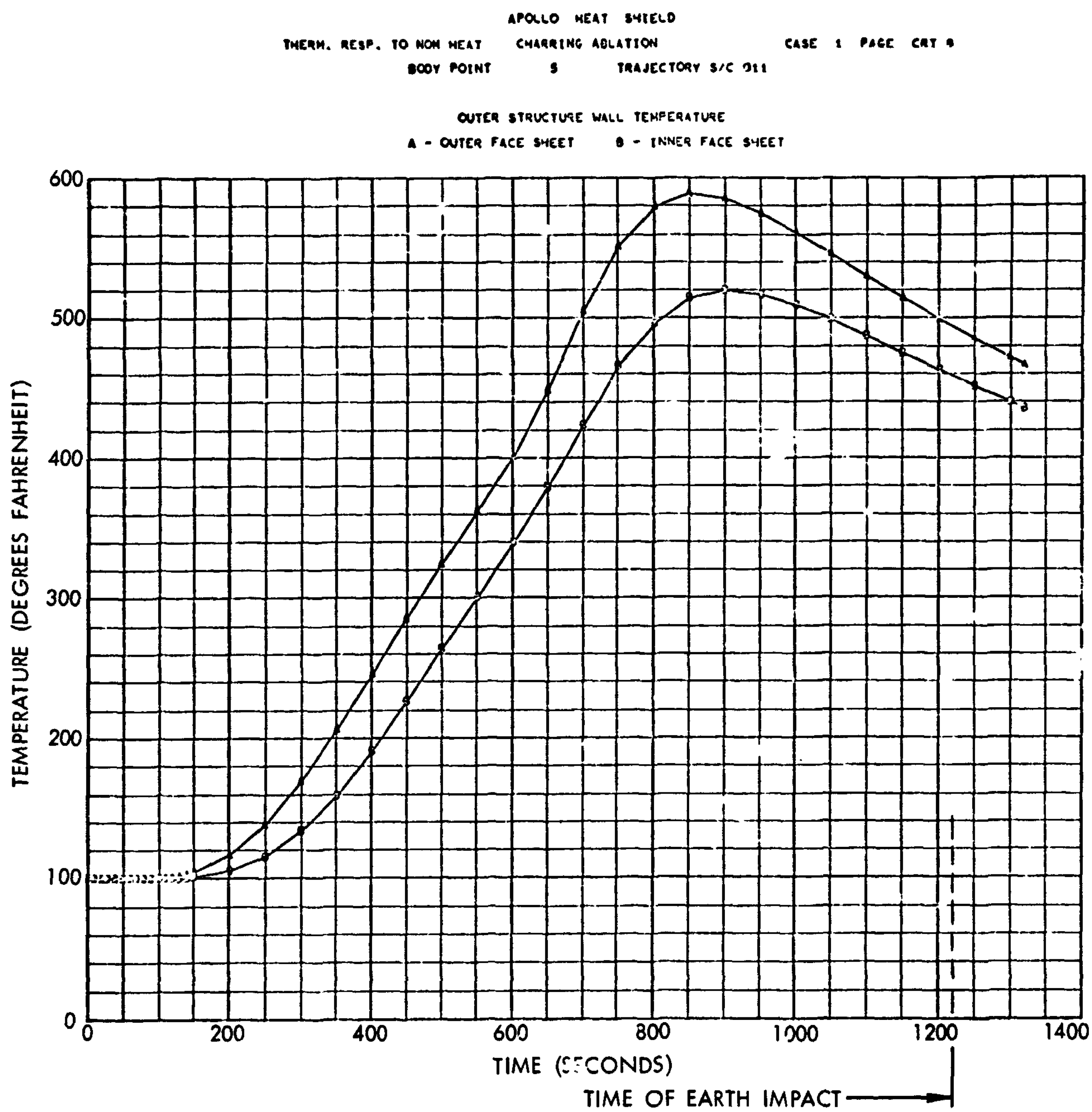


Figure 43. Bondline Temperature History, Location 210, Ablator Sized for, and Subjected to, 28,500-FPS Entry Trajectory

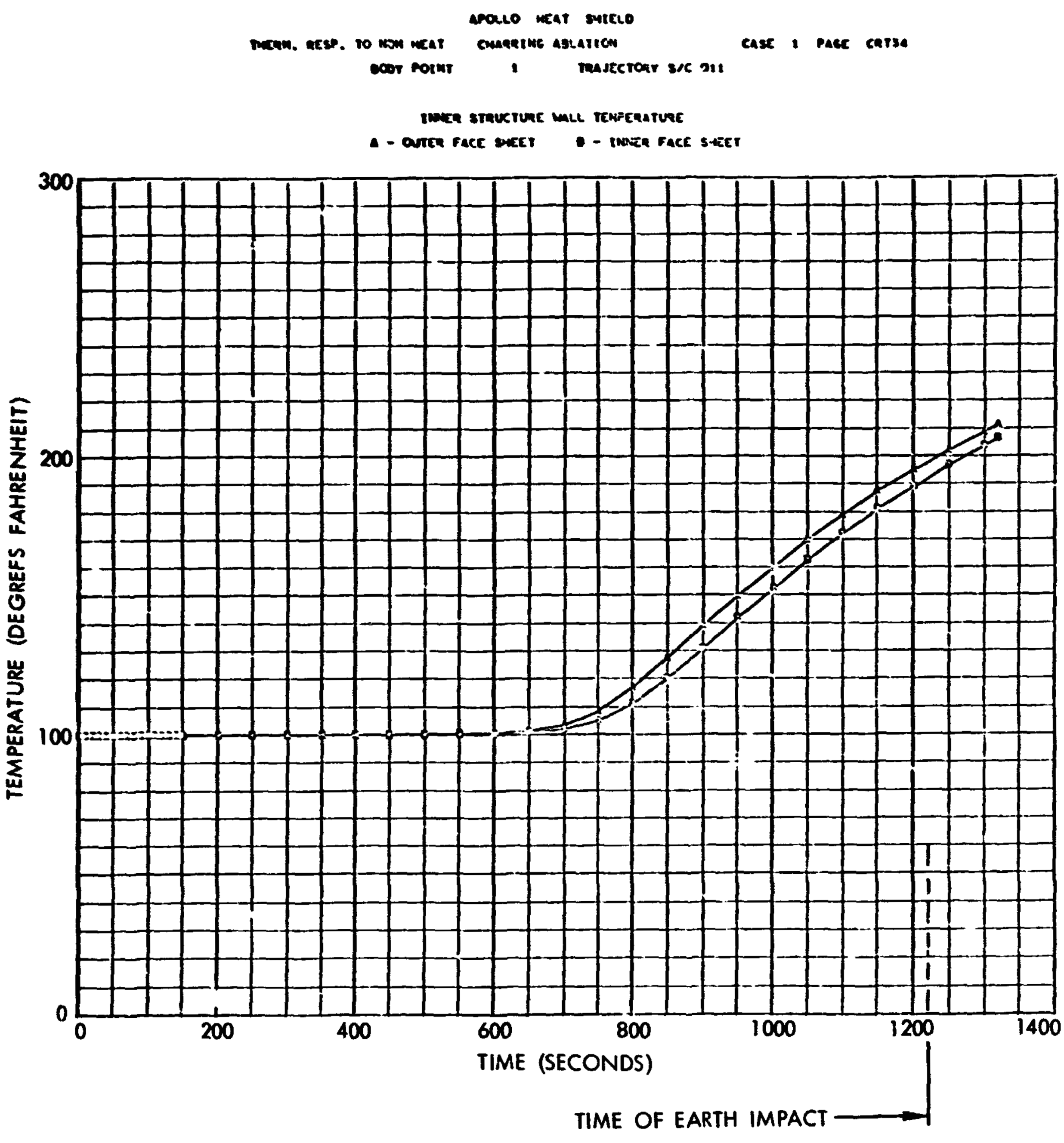


Figure 44. Aluminum Pressure Vessel Temperature History, Location 200, Ablator Sized for, and Subjected to, 28,500-FPS Entry Trajectory

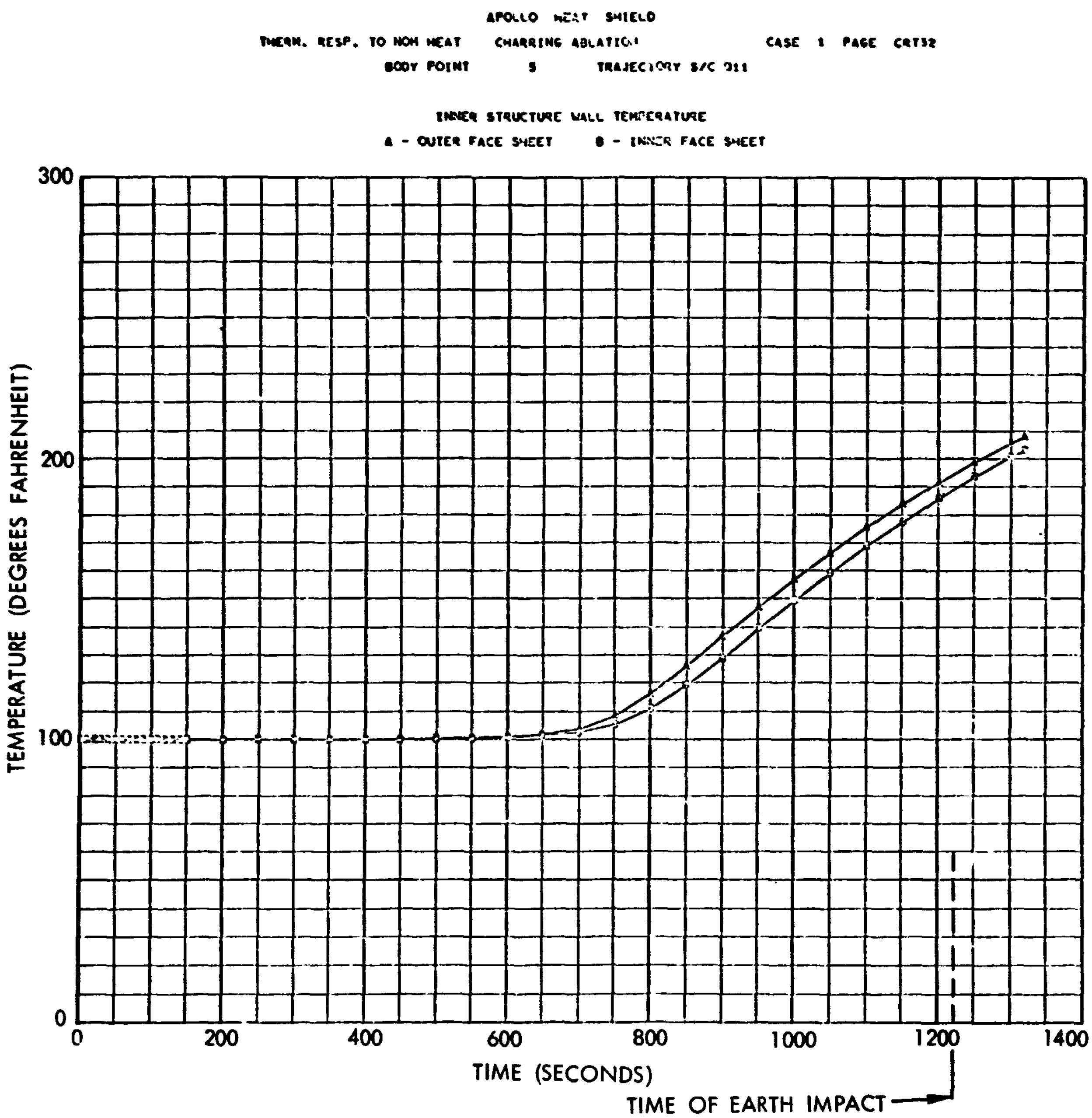


Figure 45. Aluminum Pressure Vessel Temperature History, Location 210, Ablator Sized for, and Subjected to, 28,500-FPS Entry Trajectory

renovation approach. Another advantage is time. Lead time for heat-shield renovation is approximately six months, while lead time for a new heat shield would be about two years. This difference is due to the retooling and setup time required to produce the honeycomb substrates.

SPACECRAFT 011 DATA

Heat shield degradation data from S/C 011 provided a base point for consideration. The S/C 011 aft heat shield is shown in Figure 46 as received at Downey after recovery from flight. Evident in the photo are the dolly support imprints where the salt deposits have been disturbed. This highlights an important renovation consideration: even with the sea water baked out of the porous ablator material, the remaining salt will have some effect on the aerothermodynamic performance of the material if it is to be reused. It is suspected that this will improve the performance, but this would have to be investigated rigorously before the remaining virgin or reaction zone material could be used for subsequent missions. A close examination of the photo also reveals the prevalence of Class 3 repairs and that the fabrication mode involves the assembly of gore sections.

Figure 47 is a photograph of what appears to be ablator material separation in the S/C 011 aft heat shield in the stagnation region. Upon examination it was found that the fissure was only one-half inch deep, and due to low density material in the core gape splice. This material ablated faster than that surrounding it. This condition is rare and will seldom occur. Even so, no anomalous performance was observed to have resulted from it, except that the surface recession and reaction penetration were somewhat greater in the fissure. The degree of this difference from nominal can be appreciated when it is considered that the heat rate at the bottom of a fissure with a one to two aspect (width of fissure to depth of fissure) is less than one percent of that at the surface. These anomalies, however, do have some effect on the renovation considerations, since the remaining virgin material in the vicinity of the fissure may be inadequate if the average thickness is marginal.

The fiber-glass edge members of the heat shield suffered little to no degradation on the S/C 011. Figure 48 shows an S/C 011 crew-compartment heat shield edge member in the vicinity of the stagnation region. Virtually no discoloration is evidenced beyond the char level of the ablation material.

Figure 46. Spacecraft 011 Aft Heat Shield

Figure 47. Spacecraft 011 Aft Heat Shield Ablator Separation

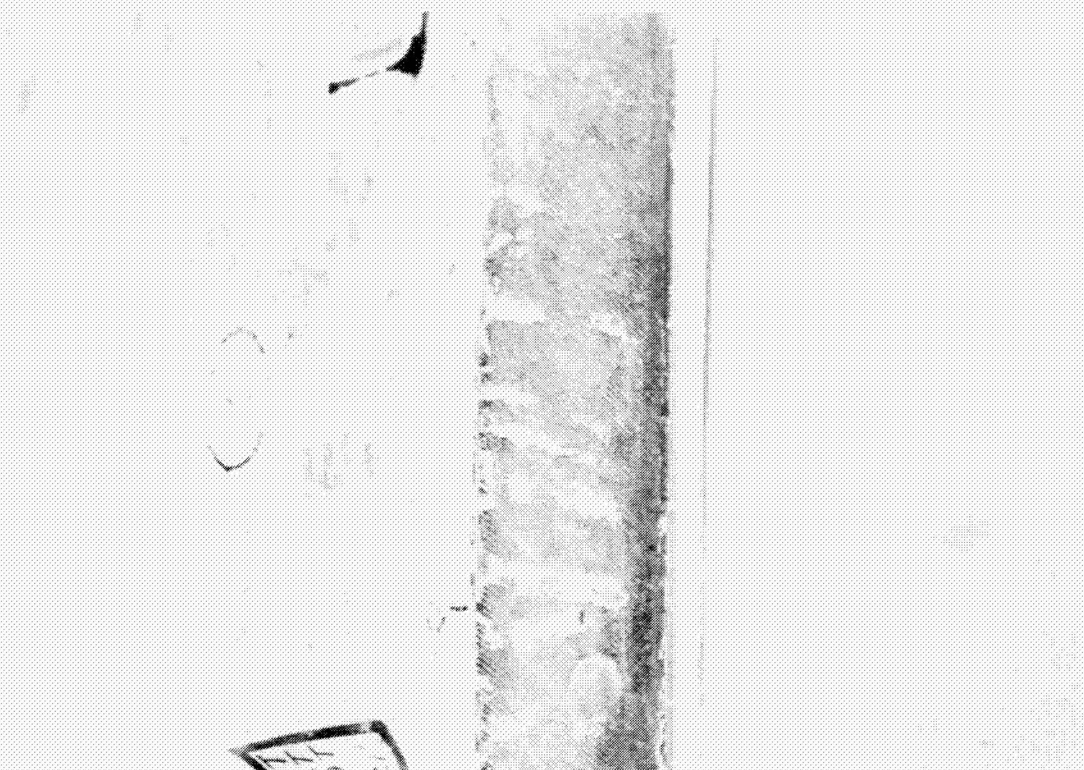


Figure 48. Spacecraft 011 Crew Compartment Heat Shield Edge Member in the Stagnation Region

V. ENVIRONMENTAL CONTROL AND LIFE SUPPORT

Components of the environmental control and life support systems (ECS/LSS) are located in both the command module and service module. All SM equipment items are lost during separation. These consist mainly of the space radiators plus their flow-control system. The remaining items, located in the CM, therefore, were studied.

OPERATIONAL DEGRADATION FACTORS

During preliminary evaluation of the various sources of equipment degradation, it was decided to delete the effects of radiation and meteoroid environments from detailed analysis for the following reasons:

1. The equipment inside the CM will be adequately protected from radiation.

2. Meteoroids can affect the ECS/LSS only during catastrophic penetration, in which case the CM would not be renovated.

To reduce the data available into usable criteria for evaluation of the degradation experienced by the ECS/LSS during a lunar mission, six primary parameters were chosen: temperature, pressure, shock, corrosive contaminants, acceleration, and life requirements.

Table 15 lists these parameters and gives the design and test requirements versus the study requirements. In the following sections, each parameter is evaluated.

Temperature

The most crucial area of potential temperature degradation was reentry heating. The estimated maximum temperature was only 126 F. Each component of the ECS/LSS was given a two-hour test at 150 F during qualification testing. During mission life testing the assembly was exposed for two and one-half hours at a CM wall temperature of 150 F. The indication here would lead to the assumption that since qualification testing exceeds both the temperature level and time of exposure, the performance degradation due to temperature exposure should be minimized.

Table 15. Evaluated ECLSS Parameters

Parameter	Design Requirement	Qualification Test Requirement	Study Results
Temperature	Storage -25 F to 105 F for 3 years Air Transportation -20 F to 140 F for 8 hours Operating 50 F ground cooling 60 F minimum space flight 150 F maximum average during entry 200 F maximum local at entry	2-hour soak at 150 F for each component 2.5 hours at 150 on assemblies 2-hour soak at 60 F	15 F CM interior, minimum 126 F maximum at reentry
Pressure	Normal 5 psia - 100% O_2 Emergency 1×10^{-4}mm Hg for 96 hours	430 hours at 5 psia life test 200 hours at 1×10^{-4}mm Hg life test	100 hours at 1×10^{-4}mm Hg
Shock	78 g's with leakage not to exceed 10 cc/hr Post landing vent valve and blower must operate	78 g's shock test	No more than 20 g's
Corrosive contaminants	Environment of 95±5% RH 95±5% O_2 1% salt fog	Submerged in 2-1/2 feet of sea water for 15 minutes; combined O_2, humidity, and salt-spray test	
Acceleration	20 g's maximum	20 g's test	Not more than 20 g's
Life Requirements	Dynamic components 1,200 hours Static components 3,000 hours	863 hours mission life test with limited maintenance	520 hours

Pressure

During mission life tests all equipment will be exposed to twice the length of time at 5 psia and 1×10^{-4}mm Hg than will be experienced during an Apollo lunar mission. The same conclusion could be drawn with respect to degradation in performance caused by vacuum exposure.

Shock and Acceleration

The estimate shock and acceleration load on equipment will not be greater than 20 g during an earth landing. The design and test value of 78 g for all assemblies will result in equipment that can withstand 20 g with little more than increased leakage or loss of calibration tolerances.

Corrosive Contaminants

The ECS/LSS equipment in the CM has been designed and tested to withstand corrosion resulting from the normally expected Apollo atmosphere. As a conservative approach it is planned to replace all items exposed to sea water or urine. Thus, this source of corrosion need not be considered.

Life Requirements

As can be seen by examining Table 15, the ECS is life-tested almost to twice the anticipated Apollo lunar mission. The most crucial components, of course, are the rotating members of pumps, fans, and blowers.

To sum up results of Table 15, it can be concluded that the ECS/LSS is designed and tested in excess of a single Apollo lunar mission; with proper renovation, this equipment contains enough reserve life for reuse in an RCM spacecraft or laboratory.

ECS STATUS DUE TO FLIGHT AND POSTRECOVERY OPERATIONS

In general terms, the command module ECS is composed of four fluid circuits and controls, which provide the required function. The fluid circuits are two oxygen, pressurization, and suit loops; water collection and distribution; and coolant circuit. Within these circuits the various methods of circulation are employed from active pumping to flow by pressure differential.

To establish some indication of the subsystem and component status, and from this status define the general renovation requirements, the types of hardware contained in the ECS must be identified in order of effect on system performance or, in this case, system degradation. By utilizing available information and data indicated by installed redundancy, component development

history, and S/C 009 postflight data, Table 16 was constructed to show the four major categories and the types of hardware in each category and arranged in the order of decreasing effect on system performance.

Table 16. Categories of Hardware in Order of Effect on System Performance

Category	Type of Component
1. Rotating machinery	Fans, compressors, pumps (all types)
2. Active machinery	Pressure regulators, modulating control, valves
3. Semi-active machinery	Solenoid and motor driven valves, check valves, hand valves, relief valves
4. Passive machinery	Sensors, electronics control boxes, flow limiters, tankage, heat exchangers, S/C plumbing and ducting, filters

SUBSYSTEM STATUS AND ASSOCIATED PERFORMANCE DEGRADATION

Rotating Machinery

The critical rotating machinery in the CM ECS is the coolant pumps, cabin recirculation fans, and suit compressors. The two areas in the motor and impeller combination subject to degradation or failure are bearings and motor insulation. In the case of motor insulation, tests can be conducted to establish the status of this parameter and associated degradation. Bearings, on the other hand, present a different problem: the exact status cannot be determined by testing in the assembled condition; disassemble and inspection are necessary to determine actual status. In the past, many companies have expended much effort to establish a method for determining or predicting bearing life with any degree of accuracy, which indicates that bearing status and associated component life is somewhat in question. However, noting S/C 009 post flight data entries for the water glycol pump assembly and the cabin recirculating fans in Table 17, indications are that flight environments and operating times caused little performance degradation, which could mean that flight environments such as those of S/C 009 will not degrade the water

Table 17. S/C 009 Post Flight Data

P/N & Name	Significant Test Parameter	Spec Value	Test Results			Remarks & Comments
			Test Value	Accept	Reject	
848020-3 Environmental Control	External leakage/ potable H_2O circuit	0.060 cc max in 15 min	0.056 cc in 15 min	x		Impact and post recovery operations, no major effects
	External leakage/ waste water circuit	4.65 cc max in 10 min	0.024 cc in 10 min	x		
	External leakage W/G circuit	0.624 cc max in 15 min	0 leakage	X		
	External leakage/ 20 psi circuit	0.176 cc max	0.824		X	Probable cause, impact g load
	External leakage suit circuit	101 cc max	346 cc		X	Probable cause, impact g load
	External leakage 100-psi circuit	334.6 SCCM	718 SCCM		X	Probable cause, impact g load
	ECU electrical continuity					Approximately 198 pins checked; 189 passed, 9 rejected
850024-3-1 W/G pump assembly (S/N 104-101)	External leakage	0.108 CU max in 15 min	3.99 cc in 7 min		X	
	Torque	5.0 in-lb max	3.5 in-lb	x		Torque, power, and flow indicate good mechanical condition
	Pressure rise	29.5 psi min	29.2 & 29.3 psi		x	Degradation insignificant to system performance
	Power consumption	36 watts max	System test 33 watts, pump 1 & 2 ATP test 50.8 & 56.4 watts	(X)	X	Estimate initial pressurization 30.5 psi at 200 lb/hr. ATP test power factor not included: PF ≈ +0.598
820918-1 Glycol temperature sensor (S/N 44-110)	Resistance	1787-1857 ohms	1761 ohms		x	

Table 17. S/C 009 Post Flight Data (Cont)

P/N & Name	Significant Test Parameter	Spec Value	Test Results: Test Value	Accept	Reject	Remarks & Comments
828550-1 O_2 Demand regulator S/N 94-104	Leakage at vent actuation	10 SSCM	16 SSCM Slow to respond		x x	Probable cause, impact g loading Probable cause, impact g loading
828290-2-1 Back pack supply valve S/N 94-106	All			x		
8258510-1-2 Emergency in-flow control valve S/N 104-104	Aneroid relief valve Calibration check	Pressure at flow of 207 SCCM to be 4.1 to 5.1 psi	Pressure at 207 SCCM 6.25 psi		x	Probable cause, impact; recalibration and adjustment required
827330-1-2 O_2 Pressure regulator assembly S/N 14-105	Leakage			x		
828260-1-2 O_2 Check valve	All			x		
810340-2-1 Tank pressure regulator	All			x		
837016-1 O_2 Pressure transducer	All			x		
848050-2-1 Water panel S/N 94-104	Visual & external leakage			x		No indication of structural damage
Cabin recirculating fans	Visual & functional			x x		No indication of structural damage Met all requirements
Cabin pressure relief	Leakage functional boost flow/ΔP	Flow below 5.5		x x		Met all requirements Met all requirements
820918-2 Glycol temp sensor S/N 104-119	All			X		
820908-2-1 Back pressure control S/N (84-107) (85-107)	Electrical			X		

Table 17. S/C 009 Post Flight Data (Cont)

P/N & Name	Significant Test Parameter	Spec Value	Test Results			Remarks & Comments
			Test Value	Accept	Reject	
812000-4-1 Glycol evaporator	External leakage	1.0 SCC in 15 min	Approximately 100 cc/min		x	Leakage at mounting flanges
	Flow vs P	3.4 lb/min at 1.53 in Hg max	3.4 lb/min at 0.68 in Hg	x		
828390-1-2 Back pressure control valve	Electrical, mechanical, and hydraulic			x		
82?920-1 Glycol temperature sensor S/N 14-10?	All parameters			x		
820910-1-1 Wetness control S/N 14-104	All			x		
836106-1 Back pressure sensor 1 S/N 7884	Electrical			x		
828470-1 Glycol evaporator H_2O control valve S/N	All			x		
8360?6-1-1 Temp transducer power supply S/N 14-107	Excitation output voltage	2.7±.02 vrms	2.730		x	Recalibration required
836056-1-1 Signal amplifier S/N 14121	Output voltage at percentage of range	75% 3.75±.025 100% 5.0±.025 50% 2.5±.025 25% 1.25±.025 0% 0.±.025	3.787 5.047 1.267 2.50 +.027		x	Calibration required could be caused by normal use & vibration environment
836056-2 & -1 S/N 14-129 S/N 14-131 S/N 14-130	 Calibration Calibration Calibration	 All parameters All parameters All parameters		 x x x		
84830-2-1 O_2 Valve panel S/N 124-104	All			x		
810200-2 Cabin pressure regulator S/U 14-102	Flow rate	0.019 to 0.022 lb/min	0.0224 lb/min		x	Probable cause, impact g loading

glycol pumps and cabin recirculating fans beyond renovation. Since impact loading and boost vibration appear to cause bearing problems, similar data from S/C 009 would be expected for all other S/C's due to similiarities in trajectory.

There are no data in Table 17 for the suit compressor, since S/C 009, being unmanned, had no requirement for suit compressor.

Included in the rotating category are the cyclic accumulators, which are oxygen-pressure actuated. Development history and flight test operations have indicated that the component is subject to leakage due to material fatigue over extended operating periods. It would be expected that the combination of operation time and temperature exposure would cause leakage. In addition, the component might show contamination from being exposed to perspiration.

Active Machinery

ECS pressure regulators, including cabin-pressure relief and oxygen-demand regulators, can be grouped for this discussion. The causes of failure are contamination and failure of springs, diaphragms, and seals. Regulators illustrate somewhat the same unknowns as motor bearings in that disassembly is required to evaluate status to some degree of accuracy. Pressure regulators operating at high pressures and controlling to relatively wide pressure tolerances are not expected to show much degradation in performance due to rugged internal construction. The g loading of impact may cause leakage and, in some cases, necessitate recalibration and adjustment.

From Table 17 it can be seen that the higher pressure regulators, like the tank pressure regulator and the O_2 pressure regulator assemblies, could pass ATP requirements after flight. On the other hand, the low-pressure oxygen in-flow valve and the cabin-pressure regulator would require recalibration or repair prior to reuse.

The oxygen demand regulator is expected to suffer some damage from impact g landing. Regulators of this type must be delicate in order to perform the necessary function at very low pressure differentials (approximately 9 inches of water to 4.0 psi). In addition, normal usage requires that suit-circuit oxygen in the three-man suit condition be vented through the relief port. This is a possible source of contamination, the degree of which is established by suited operation and the number of times the relief function is required.

One of the most critical components in the pressure regulation system is the cabin-pressure relief valve. Due to the valve's exposure to vacuum during flight, hot incoming gases during entry, and salt water after impact, degradation in performance, leakage, and, to some degree, internal and external corrosion are expected.

Table 17 indicates that the relief valve did not show any degradation in performance in the way of leakage and boost relief, but corrosion on external parts and some deviation in entry pressurization were observed. This would also be expected since S/C 009 did not experience the extended vacuum exposure of flight and the higher temperature conditions of lunar entry.

The flow-modulating valves in the water-glycol loop are the cabin temperature control valve and the glycol temperature control valve across the radiator system. These components will see many operating cycles throughout the flight portion of the lunar mission and are subject to wear of moving parts, internal leakage, and possible water-glycol contamination. External leakage is also expected as a result of impact g loading.

The suit circuit bypass valve is expected to show similar effects as the low-pressure regulator since it also operates on a very low pressure differential. This component is expected to require adjustment as result of impact loading. In addition, the suit bypass valve is exposed to suit-circuit atmosphere during suited operation and, as such, could be expected to show contamination. Also, as is the case of all suit-circuit components, high temperature and vacuum extremes do not occur, and very little performance degradation is expected as a result of these environments.

Semi-active Machinery

This category of hardware is rugged and under normal mission conditions operates little; therefore, it would not be expected to be degraded from wear or mission life. Some components in this category are located on structure in the CM that can be expected to reach relative high temperatures (200 F maximum) concurrent with impact. The combination of temperature and impact would cause deformation of seals and seats; therefore, internal leakage is expected. External leakage is not expected to be a major factor in this type of hardware except for components that use gaskets.

Data shown in Table 17 under Back Pack Supply Valve, O_2 Check Valve, and Water Panel indicate the above status. However, it should be noted that components were subjected to the maximum temperature environment at impact.

As mentioned previously, the suit-circuit components are not expected to show degradation from temperature and vacuum extremes. The suit-circuit check valves are expected to show contamination as a result of exposure to suit-circuit atmosphere.

Passive Machinery

With the exception of sensors and electronic boxes, these components are rugged and have no moving parts to cause wearing problems. In general, temperature and vacuum exposure should cause no degradation, with the exception of water tanks, which will be exposed to high temperature during entry and to salt water after impact. These conditions would result in tank bladder damage and external corrosion and would require replacement of the entire tank. Both the suit-circuit heat exchanger and the glycol boiler contain metal wicking that could be expected to show contamination and corrosion after being subjected to perspiration from the crew. In addition, the metal wick might be damaged on impact. Also, it should be expected that impact g loading could also cause external leakage.

Table 17 indicates that external leakage existed and that internal flow paths had been damaged, causing pressure to continue to drop.

The considerations of leakage would also apply to the cabin heat exchanger. However, as can be seen from Table 17, the cabin heat exchanger survived the S/C 009 flight environment without any functional degradation.

Spacecraft plumbing and ducting would be expected to show some leakage as a result of impact. The extent of leakage could be established by test and damage-repaired; however, it has been generally decided to replace all plumbing and ducting during renovation. The data on Table 17 for the ECU give some indication of the leakage that can be expected and show little danger of secondary damage.

General Conclusions from S/C 009 Postflight Analysis

The S/C 009 ECS was in relatively good working order, as illustrated by the types of rejections shown in the postflight acceptance and system testing report (AiResearch Report No. SS-1917-R dated 13 July 1966). The major rejections are on the basis of flow, leakage, and calibration. These conditions are attributed to the impact g loading, since the flight environments of temperature and pressure and mission duration did not exceed the design requirements of the ECS hardware.

Specific conclusions are shown in Table 17 under Remarks and Comments.

RENOVATION REQUIREMENTS

Renovation requirements for individual equipment items are listed in Table 18.

Since all of the part numbers for Block I and II are not the same, the table has been divided into three lists as follows:

List No. 1. All equipment items common to both Block I and Block II

List No. 2. Additional equipment items applicable to Block I spacecraft (List No. 2 plus List No, 1 will contain all items on Block I spacecraft).

List No. 3. Additional equipment items applicable to Block II spacecraft (List No. 3 plus List No. 1 will contain all items on Block II spacecraft).

Part Numbers

The numbers in the Part Number column of Table 18 fall into three categories:

All 800,000 series numbers are AiResearch part numbers.
All number prefixed "ME" or "V" are NAA part numbers or applicable drawing numbers.
In some cases, an item will have both an AiResearch and an NAA part number.

Refurbishment Description Coding

After study of the refurbishment requirements of the various types of equipment items, it was found that the various types of activities could be placed in four categories, requirements consisting of any one or combination thereof:

Code 1. Disassemble, inspect, and clean all parts; reassemble, incorporating new seals, jackets, diaphragms, flapper valves, and springs.

Code 2. Check calibration and/or recalibrate to obtain operation and/or control at proper tolerances for temperature, pressures, flows, etc.

Table 18. ECS/LSS Renovation Requirements

LIST NO. 1

Component			Test & Reuse	Replace	Refurbished	
Item No.	Description	*Part No.			Description	Percentage of Original Cost
e	ECU main package	848020-1 ME901-0216				
1 .5	CO_2 and odor absorber assembly	811400-2			1	15
	Check valve				1	15
	Manual selector valve				1	15
4.16 A B	Demand pressure and relief valve Manual selector valve Pressure regulator and relief valve Pressure regulator and relief valve	827300-1			1, 2	20
1.3	Suit circuit return air check Valve assembly check valve	850022-1			1, 2	15
1.8	Debris trap	811100-2			1	15
1.10 A	Suit compressor Compressor No. 1 Compressor No. 1 motor	826000-2			1, 3	20
B	Compressor No. 2 Compressor No. 2 motor				1, 3	20
1.11	Check valve Check valve	850137 850138			1, 2 1, 2	15
1.16	Suit bypass valve	810520			1, 2	15
1.36 A B	Cyclic accumulator valve assembly Cyclic accumulator valve assembly	828020-1			1, 4 1, 4	15
1.38 A B	Cyclic accumulator control Cyclic accumulator control	820914			2 2	5 5

Table 18. ECS/LSS Renovation Requirements

Component			Test & Reuse	Replace	Refurbished	
Item No.	Description	●Part No.			Description	Percentage of Original Cost
1.39	Cyclic accumulator O_2 warning Sensor	820928-1			1, 2	15
1.40	Signal conditioner	820930-1			2	5
1.46	Diverter valve	829160-1			1	15
1.51	Shutoff valve (waste H_2O)				1, 4	15
2.5 A	Check valve from 1.29				1, 2	15
2.6	Glycol evaporator	812000-2		X		
2.22	Glycol temperature control	820010-1			2	5
1.41	Temperature sensor	820960-1			1, 2, 4	15
1.43	Back pressure control valve	828870-1			1, 2, 4	15
2.28 A B C	Manual valve from 1.29 Manual valve (bypass) Manual valve, glycol pump to reservoir	827050			1, 4 1, 4 1, 4	15
2.29	Glycol reservoir	812800-1			1, 4	15
2.39 A	Back pressure control valve (primary)	820928-1-1			1, 2, 4	15
2.40 A	Back pressure temperature control (primary)	820956-2-1			2	5
2.42	Glycol temperature control valve	828540-3-2			1, 2, 4	20
2.45	Glycol temperature sensor	820980-2-1			1, 2, 4	15
2.47 A D	Glycol temperature sensor (primary) Glycol temperature sensor (primary)	820980-4-1			1, 2, 4 1, 2, 4	15

Component			Test & Reuse	Replace	Refurbished	
Item No.	Description	*Part No.			Description	Percentage of Original Cost
2.49 A	Temperature sensor (primary)	820972-1-1			1, 2, 4	15
2.50 A	Glycol temperature sensor (primary)	820980-4-1			1, 2, 4	15
5.14	Wat r chiller	812400-2			4, Leakage test	5
7.6	Pressure transducer, compressør inlet	837032			2	5
7.12	Differential pressure transducer, suit compressor	837054-1			2	5
8.1A	Pressure transducer, glycol pump-cut (primary)	837008			1, 2, 4	15
8.16 A	Pressure transducer, glycol accumulator quantity measurement (primary)	837026			1, 2, 4	15
8.24 A	Steam duct pressure switch (primary)	836114-1-1			2	5
12.1	Power supply - temperature transducers	836066-1			2	5
	Oxygen valve panel	848030-1 ME284-0144				
4.24 A B C D E	O_2 pressure regulator assembly Pressure relief valve Pressure relief valve Manual selector valve Check valve Check valve	827330-1			1, 2	20
4.22 A B C	Emergency O_2 in-flow control valve Pressure regulator Pressure regulator Manual valve	827550-1			1, 2	20

Table 18. ECS/LSS Renovation Requirements (Cont)

Component			Test & Reuse	Replace	Refurbished	
Item No.	Description	*Part No.			Description	Percentage of Original Cost
3.28	Cabin pressure regulator	810200-1				
A	Pressure regulator				1, 2	20
B	Pressure regulator					
C	Manual valve					
5.24	Tank pressure control & relief valve	810340				
A	Manual valve				1, 2	15
B	Manual valve					
C	Pressure relief valve					
D	Pressure relief valve					
4.25 A	Check valve	828260-1			1, 2	15
9.2	Flow transducer, oxygen supply	837028			2	5
9.8	Outlet pressure transducer, oxygen supply regulator	837016			2	5
3.1	Cabin pressure relief valve	810400-1		X		
A	Pressure relief valve with manual override					
5.22	Potable water supply assembly	812900-1			1	15
	Manual valve	ME901-0223				
	Heater					
1.31	Suit flow limiter	ME901-0220	X			
A			X			
B			X			
C			X			
D			X			
1.32	Suit hose connector assembly	810800				
	Check valve					
	Connection - supply	ME414-0242			1, 2	15
	Connection - return					
	Shutoff valve					

Table 18. ECS/LSS Renovation Requirements (Cont)

Component			Test & Reuse	Replace	Refurbished	
Item No.	Description	*Part No.			Description	Percentage of Original Cost
1.33	Suit hose connector assembly Check valve Connection - supply Connection - return Shutoff valve	810804 ME414-0243			1, 2	15
1.34	Suit hose connector assembly 810802 Check valve Connection - supply Connection - return	ME414-0231			1, 2	15
None	Water panel	848050-1 ME284-0143				
5.2 A B C D E	Check valve Check valve Check valve Check valve Check valve	827360-2			1, 2 1, 2 1, 2 1, 2 1, 2	15
5.18 A B C D	Shutoff valve	827410-1 ME284-0154			1 1 1 1	15
5.19	Water tank pressure relief valve	827970-1			1, 2	15
5.23	Water pressure relief valve Water pressure relief valve Manual selector	827830-1			1, 2 1, 2 1	15
None	Cabin recirculation unit	848060 ME901-0217				
2.5 B C	Check valve Check valve				1, 2 1, 2	15

Table 18. ECS/LSS Renovation Requirements (Cont)

Component			Test & Reuse	Replace	Refurbished	
Item No.	Description	*Part No.			Description	Percentage of Original Cost
2.13 A B	Cabin temperature control valve Cabin temperature control valve	850028			2 2	5
3.2	Cabin heat exchanger	812100-2			4, Leakage check	10
4.34 A B	Oxygen selector and check valve Manual valve Check valve				1 (Entire value)	15
1.14	CO_2 and odor absorber removable Container (26 replacements)	811450-1 ME901-0218		X		
1.17	Suit compressor selector switch		X			
1.48	Switch, diverter valve (1.46)		X			
2.16	Glycol pump selector switch		X			
2.28 A B C D	Glycol shutoff valve Glycol shutoff valve Glycol shutoff valve Glycol shutoff valve	828570-2			1, 4 1, 4 1, 4 1, 4	15
3.5	Cabin temperature selector					
3.6	Cabin temperature anticipator	820110-1			1, 2, 4	15
3.7	Cabin temperature control	820010-1			1	15
3.8	Cabin temperature sensor	820100-1			1	15
3.14	Cabin blower closure	848010-1	X			
3.17	Cabin blower selector switch		X			

Table 18. ECS/LSS Renovation Requirements (Cont)

Component			Test & Reuse	Replace	Refurbished	
Item No.	Description	*Part No.			Description	Percentage of Original Cost
3.18 A B	Cabin recirculating blower	826010-1			1, 2, 3 1, 2, 3	15
4.17	Manual O_2 metering valve	827310-1			1	15
4.25 B C D	Main O_2 supply check valve Check valve (PLSS entry)	828260-1			1, 2, 4 1, 2, 4 1, 2, 4	15
4.26 A B C	Manual valve to O_2 surge tank Manual valve from O_2 storage system Manual valve (PLSS entry)	ME284-0191 ME284-0191 ME284-0191			1 1 1	15
4.27	Pressure regulator & relief Valve (from O_2 surge tank)	ME284-0192			1, 2	15
4.31 A B	Oxygen system filter	ME286-0034-001			1 1	15
5.10	Potable water tank	812300-1		X		
5.15	Waste water tank	812200-1		X		
5.18 C	Manual valve to potable water tank	827410			1	15
6.1	PLV blower	826070-3		X		
7.17	Temperature sensor, gaseous, suit supply	836054-1			1, 2	15
8.17 A	Pressure transducer, steam duct (primary)	837036			1, 2	15
8.22	Temperature sensor, steam duct	836060-1		X		
8.2[illegible]	Space radiator outlet temperature transducer	836058-1			1, 2	15
9.11	CM cabin atmosphere temperature sensor	836064-1			1, 2	15

Table 18. ECS/LSS Renovation Requirements (Cont)

Component			Test & Reuse	Replace	Refurbished	
Item No.	Description	*Part No.			Description	Percentage of Original Cost
10.5	CM cabin total pressure transducer	837044			1, 2	15
11.2	Waste water tank quantity measurement pressure transducer	836010		X		
11.10	Potable water tank quantity measurement pressure transducer	836080				
13.1 A B C D E F G	Sensor amplifier Sensor amplifier Sensor amplifier Sensor amplifier Sensor amplifier Sensor amplifier Sensor amplifier	836056-1			2	5
71.0	CO_2 sensor	GFE			1, 2	15
90.1	Steam duct heater	ME363-0013-0001		X		
90.2	Urine nozzle heater	ME363-0012-0001		X		
40.3 A	Manual backup valve	ME901-0068-0001			1, 4	15
B	Battery vent valve	ME901-0068-0004		X		
40.4	Vacuum cleaner	ME901-0024-0005	X			
40.5	Urine nozzle	V16-601421			1, 4	15
40.6	Fecal canister assembly	V16-601418-501			1, 4	15
40.61	Flex hose	ME271-0015-0003		X		

Table 18. ECS/LSS Renovation Requirements (Cont)

LIST NO. 2

Component			Test & Reuse	Replace	Refurbished	
Item No.	Description	*Part No.			Description	Percentage of Original Cost
2.30	Glycol pump assembly (2.30 Block 1) Accumulator Isolation valve	850024-2				20
A B	Pump Pump				1, 3, 4 1, 3, 4	
A B	Check valve Check valve				1, 4 1, 4	
	Filter			X		
1.27	Suit evaporator temperature sensor	802050-1			1, 2	15
1.28	Suit evaporator temperature control	820010-1 ME901-0219			2	5
1.29	Suit circuit cooling unit Suit evaporator Water evaporator Glycol-To-suit air heat exchanger Water separator Water separator pump assembly	813000-1		X		
5.30	Suit evaporator water in-flow control valve	827930-1			1	15
5.29A	Water control valve (primary)	828470			1	
2.28 E F G	Glycol shutoff valve Glycol shutoff valve Glycol shutoff valve	828570-2			1, 4 1, 4 1, 4	15
6.2	PLV inlet valve	816034-1-1		X		
40.1	Blower Waste management system	ME901-0030-0005		X		

Table 18. ECS/LSS Renovation Requirements (Cont)

Component			Test & Reuse	Replace	Refurbished	
Item No.	Description	*Part No.			Description	Percentage of Original Cost
40.2	Manual selector valve	ME284-0076-0001			1, 4	15
	Blower switch (part of selector valve)				1, 4	15
40.7	Urine disposal lock	ME901-0029-0004		X		
40.8 A	Check valve	ME284-0050-0010			1, 4	15
B	Check valve	ME284-0050-0011			1, 4	15
C	Check valve	ME284-0050-0012			1, 4	.15
2.2 A B	Pressure relief valve Pressure relief valve	828450			1, 2, 4 1, 2, 4	15

Table 18. ECS/LSS Renovation Requirements (Cont)

LIST NO. 3

Component			Test & Reuse	Replace	Refurbished	
Item No.	Description	*Part No.			Description	Percentage of Original Cost
2.48	Pump package (Block II)	825070-1-2			1, 3, 4	
1.29	Suit circuit cooling unit	813410-1		X		
2.7	Glycol evaporator (see loop)	812000-2		X		
2.23A B	Glycol temperature sensor Glycol temperature sensor	820080-1			1, 2, 4 1, 2, 4	15
2.39B	Back pressure control valve (secondary)	820928-1-1			1, 2, 4	15
2.40 B	Back pressure temperature control (secondary)	82956-2-1			2	5
5.29 B	Water control valve (secondary)	828470			1	15
8.1B	Pressure transducer, glycol pump-out (secondary)	837008			1, 2, 4	15
8.16 B	Pressure transducer, glycol accumulator quantity measurement (secondary)	837026			1, 2, 4	15
8.24 B	Steam duct pressure switch (secondary)	836114-1-1			2	5
12.1	Power supply-temperature transducers	836066-1			2	5
4.28	Oxygen shutoff valve	828960-1-1			1, 2	15
2.1 A B	Water-glycol check valve Water-glycol check valve	828370-1-1			1, 2, 4 1, 2, 4	15 15
2.16	Glycol pump selector switch		X			
2.36 A B C	Diverter valve (water-glycol) Diverter valve (water-glycol) Diverter valve (water-glycol)	828950-1-1			1, 4 1, 4 1, 4	15

Table 18. ECS/LSS Renovation Requirements (Cont)

Component			Test & Reuse	Replace	Refurbished	
Item No.	Description	*Part No.			Description	Percentage of Original Cost
3.30	CM LEM pressurization valve (three-way)	828920-1-1			1	15
3.31	Pressure equalization valve	828930-1-1			1	15
3.32	LEM pressure gauge	828940-1	X			
3.33	CM hatch vent valve				1	15
6.7	PLV outlet valve	816032-1		X		15
8.17 B	Pressure transducer, steam duct (secondary)	837036			1, 2	15
2.5 B	Check valve from 1.29 (sec loop)				1, 2	15
2.35	Glycol manual metering valve (Block II)	828990-1-1			1, 4	15
2.47 B C	Glycol temperature sensor (secondary) Glycol temperature sensor (secondary)	820980-4-1 820980-4-1			1, 2, 4 1, 2, 4	15
2.49 B	Temperature sensor (secondary)	820972-1-1			1, 2, 4	15
2.50 B	Glycol temperature sensor (secondary)	820980-4-1			1, 2, 4	15

Code 3. Replace all bearings on rotating members.

Code 4. Check for corrosion caused by ethylene glycol or other operating fluid.

These code numbers identifying the refurbishment required are entered under the Description column in Table 18.

Percent of Original Cost

The figures in this column are based on assuming 10 percent for disassembly and assembly with no testing of the renovated item included.

Item Numbers

The number appearing in the Item Number column of Table 17 are those found on AiResearch schematic drawing 848149 for Block I and 848150 for Block II.

POSTRECOVERY OPERATION REQUIREMENTS AND CONSTRAINTS

The general ECS postlanding recovery procedures and constraints are concerned with the RCM ECS/LSS fluid systems and the reuse of installed plumbing and ducting. To achieve this goal, the following procedure is followed:

1. Immediately upon its recovery, the ethylene glycol circuit shall be continuously circulated by use of ground support equipment on board ship, at the forward area and during transportation to manufacturing.

2. To maintain the cold plates in a usable condition, they must be processed to the original dry-shipment condition immediately upon reaching manufacturing. This can be done by using normal GSE and following the applicable paragraphs of existing process specification MA0201-5144.

3. After the flushing, cleaning, and drying are completed per the above specification and the components to be renovated are removed, the plumbing lines should be pressurized with dry nitrogen and capped.

4. Prior to reinstallation of components, process specifications for line cleaning should be followed. Specifications are as follows: MA0206-0128 for O_2, MA0206-0174 for W/G, and MA0206-0175 for H_2O.

It should be noted that the above requirements consider the cold plates as part of installed plumbing. Apollo ECS experience has indicated the potential corrosion tendencies of cold plates and heat exchangers. Efforts to date have indicated that corrosion occurs when the coolant is allowed to stand in equipment, such as cold plates, with small volumes and large surface areas.

The primary cause here is utilization of inhibitors in the small volume of coolant, resulting in corrosion caused by noninhibited coolants. To preclude this situation, circulation should be maintained until the system can be properly cleaned and dried.

EXPLORATION TEST REQUIREMENTS

A number of tests could be run on the first suit-circuit heat exchangers and glycol evaporators to ascertain the extent of degradation of the wicking material and porous plates. These tests would consist of performance-testing and then disassembling or cutting up the first units received to obtain finite data on the extent of degradation. This procedure might result in considerable savings on subsequent units.

Further savings might be made by exploratory testing to evaluate the effect of degradation on the elastomers in semipassive items, such as hand valves, solenoid valves, check valves, and relief valves and especially backup-type check valves as used in the glycol circuit to close the line if the hand valve is not closed before separation of the CM from the SM.

Certain hand valves that do not have free internal elements subject to damage during impact loading may benefit greatly from exploratory testing. Most of these type items can stand some increase in internal leakage and still operate satisfactory in the system, but no increase in external leakage. Inspection and renovation of first items of these categories may indicate that no or little renovation is actually needed.

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VI. ELECTRIC POWER

CONDITION OF RECOVERED SYSTEM

The Apollo electric power system (EPS) and sequencing system (SS) have been investigated to determine the feasibility of reusing them in a renovated command module spacecraft (RCMS). In general, the EPS equipment should be in good condition after a flight. This, of course, assumes that the EPS has operated normally during a flight and that there has been no physical damage to the equipment. The SS control components, on the other hand, have a serious lifetime limitation and should be replaced. A summary of the results of this investigation is shown in Tables 19 and 20, along with a recommendation of each component's reusability. Included in Table 19 are some service module components which are listed to identify the equipment that may be required as new items in a renovated command module laboratory (RCML).

PHYSICAL EFFECTS OF ENVIRONMENT

The physical environment to which the EPS and SS are exposed does not significantly contribute to the degradation of the equipment. With the exception of some wiring harnesses and connectors, the EPS is contained within the crew compartment. Thus, the environment to which it is exposed is less severe than the external space environment.

The natural environments, including radiation, micrometeoroids, and vacuum, are not cause for a great deal of concern. The radiation level to which the EPS and SS will be exposed is predicted to be several orders of magnitude below that necessary to damage the equipment. Micrometeoroid damage also has not been considered too seriously as a subsystem operational problem. If a micrometeoroid should have sufficient energy to penetrate the command module and damage a piece of the EPS or SS, the command module would probably be damaged to a point where a successful reentry is questionable. The vacuum environment in the crew compartment has been estimated to be 10^{-4} mm Hg due to spacesuit leaks. This vacuum will occur during extravehicular activities and should not exceed 100 hours during a flight. The qualification tests for the equipment simulated this environment, and the results did not reveal any adverse effects.

The induced environment, such as vibration, shock, and temperature, called out in the Apollo Environment Specification (Reference 1) has been

Table 19. Electrical Power System (EPS) Components

Component	Description	Block I	Block II	Replace	Test and Reuse	Refurbish (percent)
C14A1	A-c power control	V16-451136	V36-454000			5
C14A2	D-c power control	V16-451168	V36-452020			5
C14A3	Motor switch control	V16-454352	V36-454050			5
C14A7	Main CB panel	V16-451165	V36-452050			30
C14A8	Power factor correction	V16-452534	V36-452000	X		
C14A12	Fuse box	V16-451115	V16-451115	X		
C14A14	Auxiliary d-c power control	V16-451175				5
C14A15	Auxiliary d-c CB panel	V16-451189				40
C14A16	Battery CB panel	V16-451167	V36-452200			40
C14BT1, 2, 3	Battery reentry and postlanding	ME461-0012-0002	ME461-0012-0002	X		
C14BT5, 6	Battery, pyrotechnic	ME461-0007-0002	ME461-0007-0002	X		
C14BTC1	Battery charger	ME461-0002-0001	ME461-0002-0007		X	
C14PS1, 2, 3	Inverter	ME495-0001-0004	ME495-0001-0006		X	
	Battery vent tube	ME270-0001-0001	ME270-0001-0002		X	
C15A1	RH electrical feedthrough	V16	V36-597014	X		
C15A2	LH electrical feedthrough	V16	V36-597008	X		
C15A5	Circuit utilization box	V16-442127	V36-442213			5
S14A1	Power distribution box	V17-451208	V37-451230	X		
S14A2	F/C remote control assembly	V17-451200	V37-451210	X		
S48A4, 5	Tank signal conditioners	ME901-0157	ME901-0157	X		
S48A6	H_2 valve package	ME284-0118	ME284-0291	X		
S48A7	O_2 valve package	ME284-0114	ME284-0290	X		
S48A 10, 11	Cryogenic control system		V37-444010	X		
S48AR 107, 108, 109, 110, 111, 112	Flowmeter	ME449	ME449-0015	X		
S48LV 1 2, 3, 4, 5, 6	F/C shutoff valves	ME284-0116 ME284-0117	ME284-0289	X		
S48TK 1, 2	H_2 cryogenic tank	ME282-0027	ME282-0047	X		
S48TK 3, 4	O_2 cryogenic tank	ME282-0026	ME282-0046	X		
	H_2 cryogenic tank support skirt		ME127-0039	X		

Note: S14 and S48 components are located in the service module and are not recoverable.

Table 20. Sequencing System (SS) Major Components

Component	Description	Part Number		Recommendation
		Block I	Block II	
C18A1	Master event sequence controller	ME901-0567-0008	ME901-0567-0014	Replace
C18A2	Master event sequence controller	ME901-0567-0008	ME901-0567-0014	Replace
C18A3	Lunar docking event controller		ME476-0035-0001	Replace
C18A4	Lunar docking event controller		ME476-0035-0001	Replace
C19A1	CM RCS controller			Replace
C19A2	CM RCS controller			Replace
C20A1	Earth landing system controller	ME901-0001-0019	ME901-0001-0019	Replace
C20A2	Earth landing system controller	ME901-0001-0019	ME901-0001-0019	Replace
C20A4	Pyro continuity verification	V16-540130-101	V16-540130-101	Replace
C23B2	Postlanding vent controller	V16-540170	V16-540170	Replace

very close to the data obtained from S/C 009. Vibration data from airframe 006 tests did not exceed the requirements called for in the EPS equipment design. The inverter and battery charger are potted inside the case and the components are supported and clamped by the potting material. The motor-driven switch contacts are locked firmly in place by the internal cam mechanism and are well able to withstand the vibration. Devices using the same basic mechanism as the Apollo motor switch have successfully survived shock and vibration requirements greater than those specified for the Apollo.

Shock within the command module has been ascertained to be less than the 78 g called out in the equipment design requirement. This high shock level was called out to ensure crew safety and should normally occur in the command module impact area. An inverter at Westinghouse was subjected to the 78-g shock and was still in good condition after the test. Because of the similarity of construction, both the inverter and battery charger should withstand the shock of impact with no degradation of performance.

The thermal analysis described in Section II was performed using upgraded data from S/C 009 to determine the temperature of the structure during reentry. Table 21 shows the temperature to which several EPS components will be exposed. As can be seen, the inverters are mounted in the location of greatest heat. The inverters are also the largest active heat producers that are candidates for reuse in a later flight. Data from the S/C 001 flight show that the inverter temperature never exceeded 140 F, which is well within the operating limit. Therefore, the temperature environment should not affect the operating lifetime of the reusable EPS hardware.

The wiring harnesses, in general, should be in good condition. Cable connectors are generally moistureproof and lockwired in place to the equipment. All exposed terminals are first conformally coated with a silicon primer and then potted with a silicon rubber material. The cables within the crew compartment on S/C 009 and S/C 011 appeared brand-new after their flights. The cables external to the crew compartment are exposed to a much harsher environment and must be carefully checked before use. Of course, the CM/SM umbilical harness must be replaced. The remainder of the cables in the aft compartment appeared to be in good condition. Examination of S/C 011 revealed salt deposits on some of the harnesses, and a few connectors showed signs of corrosion on their external surfaces. No corrosion was apparent on the pins of the connectors that had been disconnected, but one of them did have slight deposits of a white material on the front surface. Whether this is residue from sea water or matter that came loose after the connector had been disconnected is not known. In general, it appeared that the wiring could be used again with little refurbishment. If proper recovery techniques are used to minimize the effects of corrosion, this should be the case on subsequent flights.

Table 21. Component Support Structure Temperatures

Components	Location	Temperature (F)
C14A1	Central lower equipment bay	93
C14A2	Aft RH equipment bay	110
C14A3	Aft lower equipment bay	134
C14A7	Aft RH equipment bay	110
C14A8	Aft RH equipment bay	110
C14A12	Aft RH equipment bay	110
C14A13	Aft RH equipment bay	110
C14A16	Aft RH equipment bay	81
C14BTC1	Aft lower equipment bay	132
C14PS1	Aft lower equipment bay	135
C14PS2	Aft lower equipment bay	132
C14PS3	Aft lower equipment bay	139
C15A5	Aft RH equipment bay	110

It may be possible to refurbish the feedthrough connectors that pass the electrical signals out of the crew compartment. The feedthroughs contain pyrotechnic devices that deadface sensitive wires during reentry be separating two sets of connectors. Once actuated, the pins within the unit are exposed and are subject to the effects of sea water during postlanding operations. However, no tests have been performed to ascertain the effects of sea water or firing of the pyrotechnic device on the reliable life of the feedthrough. Costs for the additional qualification and refurbishment could easily exceed the replacement cost. Instead of performing these necessary tests, it is recommended that new units be used.

It is evident that the environment to which the EPS and SS are exposed have a minimal effect on the hardware. The stringent requirements imposed by crew safety criteria have resulted in a conservative design philosophy. The only environment that has not been included in the design and qualification testing is the postlanding environment. However, effective postlanding techniques and less stringent design criteria for a renovated command module would minimize the effect of the lack of the postlanding requirement.

DEGREE OF DEGRADATION FOR ONE MISSION

Equipment operation provides the most serious limitation to reusability. Certain items in the EPS and SS have limited lifetimes and must be replaced. The reentry and postlanding batteries are of the silver oxide zinc secondary-battery type. These units have a guaranteed charge/ discharge lifetime of six cycles and are insufficient for consideration for use in a second flight. Because the battery cells are sealed into the battery case, the unit cannot be refurbished. The pyrotechnic batteries are a silver oxide zinc primary-type battery and are not intended for reuse. Once discharged, they must be replaced. Fuses and circuit breakers are system protection devices. Even though they have passed all qualification tests for reliability and crew safety, a single unit may have been damaged or degraded by the original flight. It would be extremely difficult, perhaps impossible, to reliably locate one of these units. In order to preserve reliable system operation of subsequent flights, therefore, it is recommended that all fuses and circuit breakers be replaced during refurbishment. The fuses in the power-factor correction assembly (C14A8 and the fuse box (C14A12) are potted in place and are extremely difficult to replace. It is recommended, therefore, that the power-factor correction assembly and fuse box be replaced, especially since their design is extremely simple.

The equipment used in the sequencing systems is also life-limited. The sequencers fire the Apollo pyrotechnics; in doing so they operate their output relays at, or beyond, their contact current ratings. For this reason, the operation of the sequencers has been limited to 100-cycles, including all preflight checks. Refurbishment of the sequencers controllers is not economically feasible. The construction of the units consists of modules potted in place in a case. Replacement of the output relay modules would require an estimated 75-percent teardown of the unit. In addition, there is a strong possibility that some of the other modules and interconnecting wiring would be damaged while potting material was being dug out. For these reasons and because of the critical nature of the sequencer system operation, it is recommended that these units be replaced.

The inverters have been designed and qualified to a 1200-hour lifetime. During checkout operations, the inverters, rather than GSE power, are usually used as the a-c source. Estimates of inverter on-time have been as high as 1500 hours for a mission checkout and flight to date. No data have been directly taken on this factor, but observed operating procedures substantiate this estimation. This estimate is confused by the fact that at most only two inverters can be on at one time, and no information is available as to whether this checkout time is performed on one of the three inverters or equally shared.

Use of the same inverter on a second 14-day mission would require an inverter with a 30-day operating life. Studies performed on the Apollo Extension Study (AES) program recommended a modified inverter to increase the reliability of the unit for a 30- to 45-day mission. The major change to the inverter is to change the output transistors to a higher rated unit.

To avoid modifying the inverter for possible reuse, certain changes in preflight operation should be investigated. First, the on-time of an inverter should be logged. Second, the a-c power should be provided by GSE during the checkout and preflight operations, except during inverter checkout. Use of these techniques would appreciably increase the flight lifetime of the inverters and pinpoint the inverters that achieve virtually no on-time because of redundancy.

The battery chargers, on the other hand, are not operated continuously during checkout or flight. These units are qualified for a 1000-hour life and should have ample capability for reuse.

Motor switches also do not approach their operating lifetime limit. The design requirement is for 1000 cycles of operation, and switches of similar design have operated many thousands of times. Actual operation on the spacecraft has been estimated to be less than 100 cycles; therefore, life should be no problem with the motor switches.

POSTRECOVERY OPERATION REQUIREMENTS AND CONSTRAINTS

Postflight operating procedures and constraints can be utilized to enhance the probability of survival of the reusable EPS equipment. In general, the equipment is sealed or potted and needs no special handling. A postrecovery washdown and corrosion control would minimize the problems of sea water contamination of the wire harness in the aft compartment. In addition, procedures should be established to prevent probing of potted terminals, cutting of wires, or disturbing of the wire harness in any way during the postflight operations. Disconnecting any connectors should also be held to an absolute minimum. If a connector must be disconnected, it should immediately be placed in a plastic sack with a desiccant and sealed for protection. During postflight checkout, if a-c power is required, an a-c ground power unit should be used to preserve the operating life of the inverters.

RENOVATION REQUIREMENTS

Refurbishment of the EPS equipment should include the following operations:

1. Remove the EPS assembly from the spacecraft and examine the equipment for any signs of physical damage. Protect all connectors disconnected from the equipment by placing them in plastic bags.

2. Refurbish the repairable hardware. Table 22 summarizes the major components included in the EPS subassemblies. With the exception of the fuses and circuit breakers, these components should not require replacement. None of them are stressed or operated near their design limits and all are well-protected from the environments.

3. Do not attempt repair of the inverters and battery chargers. If a postflight inspection reveals any physical damage, the unit must be replaced.

4. Functionally test all units. If a unit does not pass its functional test, it must be repaired or replaced as described in operations 2 or 3.

5. Inspect all wiring for nicks, abrasions, cuts, scratches, or discoloration due to heat. If a wire is damaged it must be replaced. If the potting on a terminal shows signs of damage or tampering, it should be removed and the terminal inspected. Any necessary wiring revisions should also be incorporated at this time. The harnesses should not be disturbed any more than necessary. They are extremely complex and any undue disturbances will greatly complicate the renovation processes.

6. Perform a continuity check when work on the harnesses has been completed. All connectors must be checked end-to-end to ensure that there has been no degradation from the first flight or miswiring.

7. Replace equipment in spacecraft and perform normal checkout.

EXPLORATORY TEST REQUIREMENTS

It is not anticipated that special tests will be required to evaluate the condition of the EPS components. Normal functional tests will pinpoint

Table 22. EPS Assembly Components

Component		Assembly								
Description	Part Number	C15A5 V36-442213	C14A12 V16-451115	C14A8 V36-452000	C14A2 V36-452020	C14A4 V36-452050	C14A13 V36-452170	C14A16 V36-452200	C14A1 V36-454000	C14A3 V36-454050
Lighting assembly	ME181-0163-0001					1				
A-c sensing assembly	ME431-0062-0003								1	
D-c sensing assembly	ME431-0063-0003	1								
Capacitors	ME441-0023-6282			27						
Fuse	ME451-0001-0003			27						
Fuse	ME451-0006-0001		30							
Motor switch	ME452-0036-0001				2		3			
Motor switch	ME452-0036-0002									4
Motor switch	ME452-0036-0006								6	
Circuit breaker	ME454-0011-00XX					5	2	6		
Circuit breaker	ME454-0014-00XX					8		3		
Relay	ME455-0051-0002								2	
Relay	ME455-0116-0002	3								
Diode	ME479-0051-0001					8				
Diode	ME479-0093-0001								2	
Diode	ME479-0112-0005				4					
Diode	ME479-0118-0001						4			

the operational status of the equipment. The reusable EPS items primarily consist of diodes, relays, motor switches, and potted electronic packages. Motor switches and relays would be checked by continuity and operational tests which are a normal part of a functional test. Examination of the component schematics indicates that there would be no special problems in checking these items on the completely assembled component packages. Diodes and electronic packages are static solid-state devices; consequently, there is no reliable method for evaluating or predicting their degree of degradation. If these devices pass their respective functional tests with results that are within tolerance and correlate with the data that was originally obtained, it should be possible to reuse them with complete confidence.

IN-FLIGHT MAINTENANCE

The availability concept or in-flight maintenance cannot be effectively applied to the power system. The EPS was originally designed with a redundancy concept in mind. If one component in a critical circuit should fail, another one can normally be switched into place.

RECOMMENDATIONS

In summary, the EPS should be in good condition after a flight. With the exception of the batteries, the high-cost items are reusable and most NAA assemblies can be refurbished. The only items recommended for general replacement are fuses and circuit breakers, and this is for crew safety purposes rather than because of a definite operational or lifetime limitation. Examination of S/C 009 and S/C 011 wiring shows that as much as 95 percent of the wiring harnesses could be reused. The SS, on the other hand, cannot be recommended for reuse or refurbishment because the units are operated near their design limits and have been assigned a definite lifetime by the Apollo Reliability Group. They are also difficult to repair or refurbish.

The investigation has been directed toward the reuse of the EPS in a Block II RCMS. The recommendations set forth in this section are also valid for a Block I or Block II RCML. Since the equipment would be installed in a new configuration for a laboratory, the wiring would be new but the system components would remain the same. When the EPS components have passed their respective functional tests, the EPS hardware is fully capable of being used in a laboratory with the same limitations as imposed on a renovated spacecraft.

VII. COMMUNICATIONS AND DATA

SYSTEM DEGRADATION

Table 23 summarizes the probable degradation effects of various environmental and operational factors upon the Apollo CM to communications/ data equipment. These factors include exposure to vacuum, radiation, meteoroids, temperature extremes in space, launch and reentry heating, structural loading and dynamics, humidity, salt atmosphere, operational wear, and postlanding operations. An "X" in the column under the various abbreviations indicates a probable significant degradation for the designated equipment. As indicated in the table, no communications/data equipment is expected to suffer significant degradation from exposure to vacuum, radiation, temperature extremes of space, and launch phase temperatures. This is because communications/data components are either not exposed to these factors or are designed to withstand them. The spacecraft antennas are the only communications/data components exposed to meteoroid environment. Probability of damage to these antennas by meteoroid impingement is considered to be remote. If such damage occurs, the result could be catastrophic.

Degradation occurs to the VHF recovery antennas, their release mechanisms, and the HF recovery antenna as a result of operational pyro deployment. These antennas are located on the outside of the CM, and additional degradation might occur as a result of overexposure to salt atmosphere after landing. The HF recovery antenna may be further degraded if postlanding handling techniques cannot be controlled. The HF antennas on both SC 009 and 011 were broken in the recovery hoisting operation from sea to deck of vessel.

The SCIN antennas are degraded by heat during reentry (see Figure 49). The +Z SCIN is virtually incinerated and the -Z SCIN is charred to a point where the RF radiation pattern will, in all probability, be affected.

Both the S-band omniantennas and the C-band transponder antennas are designed to survive the heat of reentry. Postflight visual inspection of S/C 011 showed evidence of charring on the C-band antennas in the aft heat shield. The antennas were removed by compressing the quartz cylinder out the reverse side of heat shield. No problems were encountered in this operation. The char was diagnosed as a phenolic residue from the ablator and was removed by subjecting the quartz face of the antenna to a polishing

Table 23. CM Communications/Data Equipment Degradation Assessment

Equipment	Block	Identification	Supplier	Primary Degradation Factors										
				V	R	M	T	L	RE	S	H	SA	OW	PL
HF transceiver	II	ME478-0065-0003	Collins											
VHF/AM transceiver	II	ME478-0067-0003	Collins							X				
	I	ME478-0023-0003	Collins											
VHF recovery beacon	II	ME478-0069-0003	Collins											
	I	ME478-0023-0003	Collins							X				
S-band equipment	II	ME478-0070-0003	Collins											
	I	ME478-0026-0003	Collins											
S-band power amplifier	II	ME478-0066-0003	Collins							X			X	
	I	ME478-0020-0003	Collins							X			X	
Audio center	II	ME473-0086-0003	Collins											
	I	ME473-0021-0003	Collins											
VHF triplexer	II	ME456-0040-0001	Collins							X				
	I	ME456-0013-0013	Collins							X				
Premodulation processor	II	ME478-0068-0003	Collins											
	I	ME478-0021-0004	Collins											
Signal conditioner	II	ME901-0713-0012	Autonetics											
	I	ME901-0081-0003	Collins											
Data storage equipment	II	ME435-0035-0003	Collins							X			X	
	I	ME435-0013-0035	Collins							X			X	
PCM telemetry	II	ME901-0719-0003	Collins											
PCM telemetry No. 1	I	ME901-0083-0002	Collins											
PCM telemetry No. 2	I	ME901-0083-0102	Collins											
Central timing equipment	II	ME456-0041-0030	General Time											
	I	ME456-0006-0034												
Up-data link	II	ME470-0101-0001	Motorola											
	I	ME470-0014-0007	Motorola											
Television equipment	II	SID 65-96	GFE							X				
	I	ME901-0090-0020	RCA							X				
C-band antennas(4)	II	ME481-0005-0004(2)	AMECOM			X			X					X
	II	M -0002(1)	AMECOM			X			X					X
	II	-0005(1)	AMECOM			X			X					X
	I	-0002(1)	AMECOM			X			X					X
	I	-0004(2)	AMECOM			X			X					X
	I	-0005(1)	AMECOM			X			X					X
HF recovery	II	ME481-0049-0002	Dehaviland									X	X	X
	I	ME481-0049-0007	Dehaviland									X	X	X

Table 23. CM Communications/Data Equipment Degradation Assessment (Cont)

Equipment	Block	Identification	Supplier	Primary Degradation Factors										
				V	R	M	T	L	RE	S	H	SA	OW	PL
VHF recovery antenna No. 1 VHF recovery antenna No. 2 VHF recovery antenna No. 1 VHF recovery antenna No. 2	II II I I	V36-716600-11 V36-716600-11 V16-716120 V16-716120	NAA/LAD NAA/LAD NAA/LAD NAA/LAD									X X X X	X X X X	
S-band omniantennas(4)		ME481-0048-0001	AMECOM			X			X					X
Antenna switch, nonlatching, 2 KMC	II I	ME452-0052-0111 ME452-0052-0002	TRANSCO TRANSCO											
Antenna switch, manual, VHF	II I	ME452-0054-0003 ME452-0052-0012	TRANSCO TRANSCO											
Video and RF connectors and cables	II I	3V34-971101 V14-970112	D/Cannon Misc						X X		X X	X X		X X
Release assembly VHF recovery antennas(2)	II	V36-716645	NAA/LAD										X	
C-band transponder	II I	ME478-0027-0003 ME478-0027-0003	Collins Collins							X X			X X	
Flight qual recorder	II I	ME435-0026-0003 ME435-0026-0002	Leach Leach							X X			X X	
VHF/FM transmitter and HF transceiver	I	ME478-022-0004	Collins							X				
Relay assembly, magnetic latching	II I	(Part of UDL) ME455-0106-0001	Fisher Akin											
Survival beacon	I	1274-90088	GFE										X	
Event conditioner	I	V16-764092-51	NAA											
SCIN antenna (-Z axis) SCIN antenna (+Z axis)	I I	V16-321610-31 V16-321377-91	NAA/SID NAA/SID						X X					

Key to Abbreviations:

V = Vacuum
R = Radiation
M = Meteoroids
T = Temperature extremes in space
L = Launch phase heating
RE = reentry
S = Structural loadings and dynamics
H = Humidity
SA = Salt atmosphere after landing
OW = Operational wear
PL = Postlanding operations

Figure 49. Spacecraft 011 Windward SCIN Antenna, +Z Location

operation using a commonly available lapping compound. It is concluded, therefore that the S-band omniantennas can be removed from the heat shield and refurbished by a simple operation. Because of relatively critical clearances however, the antennas can easily be degraded by physical damage during the operations of aft heat shield removal. The antennas may also be degraded by salt corrosion if sea water penetrates into the aft heat shield area.

The electromechanical data storage equipment is expected to be affected significantly by operational life limitations, as well as by exposure to vibration and shocks of the flight and landing. Mechanical parameters causing degradation include tape wear, bearing wear, lubricant migration resulting in dry bearings, belt wear with the subsequent possibility of breakage or slippage, braking mechanism wear due to ON-OFF cycling, and mechanical misalignment of recording heads and capstan drive elements.

The survival beacon is designed to withstand salt water immersion for a period of one hour and to then operate within specification for the life of the battery. Since this is a GFE item, no firm data are available at NAA/S&ID. It is believed that the survival beacon, if deployed, will have to be replaced. If it is not deployed, however, the batteries are the only items requiring replacement.

The RF cables located in the vicinity of the heat shield might be degraded by changes in dielectric properties resulting from heat during reentry. Also, cracks might develop in the connectors' heat-shrinkable sleeving as a result of a rapid change from space cold to reentry temperature. If cracks do occur, further degradation of cables and connectors might be caused by the humidity and salt atmosphere.

The RF and video cables will be exposed to potential damage during renovation operations. In particular, the RG-180 RF cable used on the Block I spacecrafts shows a history of failures and requires handling care in excess of normal procedures.

The following communications/data equipment units employ vacuum tubes: the VHF/AM transceiver and VHF recovery beacon of Block I; and the S-band power amplifier, TV equipment, and C-band transponder of both Block I and II. The performance quality of vacuum tubes is expected to be adversely affected by structural loading and dynamics, especially the shock of landing. It is probable that these components will be operative after CM recovery; however, confidence in their reliability is low because of mechanical stresses induced by shocks and vibration. These stresses probably will be most pronounced in the hot filament structure of the tubes. In addition, at the time of recovery, the operating time of the S-band power amplifier TWT and the C-band transponder magnetron will constitute a significant fraction of the usual life of these tubes.

The VHF triplexer and the S-band diplexer (which is a part of the S-band power amplifier equipment) are mechanical devices having critical dimension tolerances affecting frequency response of tuned cavities. Consequently, it is very probable that the cavit s will require retuning after the devices have been subjected to flight vibrations and shocks.

As previously indicated, the HF and VHF recovery antennas, located on the S/C upper deck, might be degraded by overexposure to salt atmosphere. The effect of salt corrosion on the remainder of the communications/data subsystem will depend upon the extent and nature of salt water intake into the cabin interior while the capsule is afloat in sea water. S/C 009 and 011 were equipped with sealed hatches, and it was not necessary to remove the hatch covers to permit crew egress. As a result S/C 009 and 011 cabin interiors showed no visible traces of salt water or atmosphere corrosion.

It is expected that the television equipment lens system will be reusable. However, as a result of structural loading and dynamics, some lenses might be broken, chipped, scratched, or misaligned.

POSTRECOVERY CONSTRAINTS AND REQUIREMENTS

The HF and VHF recovery antennas are located on the upper deck and are exposed to salt atmosphere. The HF antenna can also be easily damaged during the recovery operations and was broken on both S/C 009 and 011 during the hoisting of the CM. This damage was allegedly caused by rotation of the CM about the axis of rotation of the hoisting cable and rigging. Two possible means of preventing the breakage might be considered. The first, if safety measures permit, would be removal of the HF antenna by the recovery team swimmers while the CM is still in the water. Admittedly, this method has areas which require further investigation, such as the ability of the swimmers to stand on the flotation bag while performing the operation. A second approach would be the attachment of additional ropes to the hoisting rigging to prevent rotation of the CM during the hoisting operation. This method does not represent as positive an approach as might be desired, since it might be difficult to effectively man the rigging for the distances from the decks of the various Navy vessels. If the second approach proves feasible, the HF antenna could then be removed when the CM is aboard ship. Desalinization, anticorrosion preservation, and packaging for shipment to Downey would then follow.

Because of their smaller size, the VHF recovery antennas are less likely to be damaged during hoisting. However, there is a strong possibility that they will be unusable because of salt water corrosion. The VHF recovery antennas on S/C 011 were visibly corroded, probably beyond the point of refurbishment. The VHF re-overy antennas should be desalinized, preserved

with a suitable material, and possibly covered with a plastic protector. They should then stowed per the Recovery Manual Instructions. These operations should be performed immediately after the CM is hoisted aboard the recovery ship.

The exterior and interior of the DSE (Data Storage Equipment) is particularly susceptible to salt corrosion. Per present Recovery Manual instructions, this equipment is removed from the CM after the spacecraft is hoisted aboard the recovery ship. The DSE is then opened and inspected, and the magnetic tape is removed. The disassembled parts are then reinstalled, and the DSE is placed back in its normal location within the CM.

This procedure exposes the DSE differential pressure valve to a salt-laden atmosphere for the duration of the CM stay aboard ship or near the seacoast. A remedial procedure would be to purge the interior of the DSE with dry nitrogen. By replacing the moisture-laden salt atmosphere with the dry nitrogen, this procedure would inhibit corrosion. The DSE could then be packaged with desiccant material and shipped separately to Downey.

If the aft heat shield is removed on the ship, these components exposed by the removal (the RF connectors and cables, as well as the rear portions of the S-band omniantennas) should be desalinized and protected against corrosion and potential physical damage.

Appropriate means and procedures are required for the desalinization and anticorrosion preservation of the remaining communications/data equipment located within the CM.

RCM SPACECRAFT EQUIPMENT COMPLEMENT AND RENOVATION REQUIREMENTS.

Table 24 summarizes the equipment complement and renovation requirements for an RCMS. The equipment consists of the basic Block II communications/data subsystem with the exception of the C-band transponder, its antennas, and the flight qual recorder (FQR). The transponder is not required since it flew as a backup to the S-band system during the proving-in phase of the S-band development. The FQR is no longer required since this item was used for qualification testing of various hardware items during the S/C development phase.

A large majority of the communication/data subsystem should be capable of reuse with little or no refurbishment. Proof of this will be established by subjecting the designated equipment to a "pre-acceptance" type test wherein all critical functional parameters are inspected for original tolerances.

Table 24. Spacecraft Renovation Requirements - Communications/Data System

Equipment	Test and Reuse	Replace	Not Required	Refurbish: Cost	Refurbish: Description	Remarks
Data storage equipment				~$5000 for typical overhaul	Diagnostic preacceptance test; possible replacement of items such as bearings, belts, idlers, springs, seals, motors, and pressure valve; alignments; acceptance-type tests	Remaining DSE life in hours will be assessed by supplier; 30-day turn-around time.
S-band omni-antennas (4)				~$1700 (items 1–4)	1. Removal from heat shield 2. Cleaning 3. Antenna inspection and testing 4. Modification of antenna mount for thinner heat shield, if required	1. Special tools and methods required 2. Special procedure and possibly tools required 3. No comment 4. Design effort might be required
				100% of original operation cost (items 5–6)	5. Reinstallation of antennas in refurbished heat shield 6. Test of reinstalled antennas	5. No comment 6. No comment

Table 24. Spacecraft Renovation Requirements - Communications/Data System (Cont)

Equipment	Test and Reuse	Replace	Not Required	Refurbish		Remarks
				Cost	Description	
S-band power amplifier				Cost of TWT plus the same cost factor as in "Test and Reuse"	Replacement of TWT traveling wave tube	TWT supplied by Hughes Aircraft Company
TV camera equipment				Cost of vidicon plus the same cost factor as in "Test and Reuse"	Replacement of vidicon	TV camera supplied by Westinghouse
VHF antenna release mechanism (2)		X				Mechanisms are destroyed by pyro deployment
HF transceiver VHF/AM transceiver VHF recovery beacon Unified S-band equipment	X X X X					

Table 24. Spacecraft Renovation Requirements - Communications/Data System (Cont)

Equipment	Test and Reuse	Replace	Not Required	Refurbish Cost	Refurbish Description	Remarks
Audio center	X					
Premodulation processor	X					
VHF triplexer	X					
Signal conditioning equipment	X					
PCM telemetry	X					
Central timing	X					
Up-data link	X					
S-band antenna switch	X					
VHF antenna switch	X					
RF and video cables	X					
HF recovery antenna		X				Assumes that, as in the case of S/C009 and 011, the antennas will be broken during CM recovery operations. Low-cost (10-15 percent) refurbishment appears possible

Table 24. Spacecraft Renovation Requirements - Communications/Data System (Cont)

Equipment	Test and Reuse	Replace	Not Required	Refurbish		Remarks
				Cost	Description	
HF recovery antenna (cont)		X				if appropriate recovery measures are initiated. Refurbishment would consist of replacement of parts destroyed in pyro deployment, retesting, and restowing the antenna
VHF recovery antenna (2)		X				Assumes that, as in the case of S/C011, salt corrosion will be too severe to permit refurbishment. Low-cost refurbishment (10 percent) might be accomplished if recovery procedures permit timely protective measures against corrosion

Table 24. Spacecraft Renovation Requirements - Communications/Data System (Cont)

Equipment	Test and Reuse	Replace	Not Required	Refurbish		Remarks
				Cost	Description	
C-band transponder			X			Remove and place in storage
C-band antennas (4)			X		Remove from heat shield; plug holes in heat shield	Special tools and procedures required for removal
Flight qual recorder			X			Remove and place in storage

As can be seen from the table, both the HF and VHF recovery antennas are marked for replacement. New CM recovery procedures and improved techniques in the recovery operation itself will be required to prevent damage to the HF antenna. Reuse of the VHF recovery antennas depends upon the length of time they are exposed to salt atmosphere. This is an indeterminate quantity at this time. However, if timely measures can be taken, low-cost refurbishment will be possible. New storage covers, reefing cords, and pyro charges are all that will be required.

Contact with the DSE supplier disclosed that nominal overhaul of the listed parameters could be made at a cost of about $5000 per unit. Turn-around time would be about 30 days, and the supplier would assess the remaining life expectancy in operating hours of the unit.

Renovation requirements beyond those depicted in the table concern the aft heat shield. The Block II system utilizes a C-band transponder with associated antennas mounted in the aft heat shield. If the heat shield with the antennas removed is reused for future RCM spacecraft, it will be necessary to plug the mounting holes where the antennas penetrated the heat shield. An alternative course would be to leave the unused antenna in the heat shield. However, this has an undefined element of risk if the antennas are broken or cracked, as the damage might remain undetected.

INDEPENDENT RCML - BLOCK I - EQUIPMENT AND RENOVATION REQUIREMENTS

Table 25 summarizes the renovation requirements and Table 26 provides the anticipated equipment complement and physical characteristics for an independent RCML - Block I. The equipment consists of a basic Block I communications/data subsystem with the following exceptions: There are no HF or VHF recovery equipments since the laboratories are not required to reenter the earth's atmosphere. The C-band transponder and the FQR are not required because these are used only in the early S/C development phase. An additional item of equipment is required, however. This is the rendezvous radar transponder and associated antennas. This equipment is needed on rendezvous missions only.

In addition to those depicted in the table, renovation requirements for the Block I laboratory include incorporating the rendezvous radar transponder (RR/T) and antenna into the Block I RCML. This requirement also has an impact on the Block I controls and displays panel and the ECS cooling loop. The Apollo RR/T is mounted within the service module fairing on Block II vehicles and requires coldplate provisions for cooling. The RR/T antenna should be in proximity to the transponder electronics to minimize RF losses in the waveguide.

Table 25. Block I Laboratory Renovation Requirements, Communications/Data System

Equipment	Test and Reuse	Replace	Not Required	Refurbish		Remarks
				Cost	Description	
Data storage equipment				~$5000 for typical overhaul	Diagnostic preacceptance test; possible replacement of items such as bearings, belts, idlers, springs, seals, and pressure valve; alignments and acceptance type tests	Cost will rise rapidly if motors have to be replaced. The motor manufacturer is no longer in business. Remaining DSE life in hours will be assessed by the supplier; 30-day turn-around time on overhaul.
S-band power amplifier				Cost of TWT plus the same cost factor as in "Test and Reuse"	Replacement of traveling wave tube (TWT)	TWT supplied by Hughes Aircraft Company
TV camera equipment				Cost of vidicon plus same cost as in "Test and Reuse"	Replacement of vidicon	TV camera supplied by RCA

Table 25. Block I Laboratory Renovation Requirements, Communications/Data System (Cont)

Equipment	Test and Reuse	Replace	Not Required	Refurbish		Remarks
				Cost	Description	
SCIN antennas (2)		X (2)				One antenna is completely destroyed due to reentry heat. The other is badly charred. At present, it is not known whether refurbishment of the charred SCIN antenna is possible.
HF recovery antenna			X			
VHF recovery antennas (2)			X			
C-band transponder			X			
C-band antennas			X			
Flight qual recorder			X			

Table 25. Block I Laboratory Renovation Requirements, Communications/Data System (Cont)

Equipment	Test and Reuse	Replace	Not Required	Refurbish		Remarks
				Cost	Description	
HF transceiver and VHF/FM transmitter			X			
VHF/AM transceiver and VHF recovery beacon				$500 plus the same cost factor as in "Test and Reuse"	Replacement of vacuum tubes in VHF/AM unit	Both units are mounted on the same chassis. The recovery beacon portion is not required for the laboratory configuration.
Unified S-band equipment	X					
Premodulation processor	X					
Audio center	X					
VHF multiplexer	X					
Signal conditioning equipment	X					
PCM telemetry No. 1	X					
PCM telemetry No. 2	X					

Table 25. Block I Laboratory Renovation Requirements, Communications/Data System (Cont)

Equipment	Test and Reuse	Replace	Not Required	Refurbish		Remarks
				Cost	Description	
Central timing equipment	X					
Up-data link	X					
Relay assembly	X					
Magnetic latching	X					
S-band antenna switch	X					
VHF antenna switch	X					
RF and video cables	X					

Table 26. Independent RCML, Block I, Anticipated Communications/Data System

Equipment	Weight (lb)	Dimensions (in.)			Location	Remarks
		Height	Width	Depth		
VHF/AM transceiver VHF recovery beacon	17.5 Total	4.15 5.75	10.3 10.3	11.75 7.5	Internal (could be external if sheltered in appropriate environment)	VHF recovery beacon portion of this assembly not required for laboratory application
Unified S-band equipment	30	5.75	10.3	24	Internal (could be external if sheltered in appropriate environment)	
S-band power amplifier	13	5.75	10.3	7.5	Internal (could be external if sheltered in appropriate environment)	
Audio center	10	5.79	10.3	7	Internal (could be external if sheltered in appropriate environment)	
Signal conditioning equipment	47	4.1	15.5	14.5	Internal (could be external if sheltered in appropriate environment)	
PCM telemetry No. 1 PCM telemetry No. 2	45 Total	4.5 10.3	9.4 6.6	10.5 16.5	Internal (could be external if sheltered in appropriate environment)	
Data storage equipment	38	9.5	20	5.4	Internal (could be external if sheltered in appropriate environment)	Definite need not established
Premodulation processor	14.5	5.8	10.3	13.5	Internal (could be external if sheltered in appropriate environment)	

Table 26. Independent RCML, Block I, Anticipated Communications/Data System (Cont)

Equipment	Weight (lb)	Dimensions (in.)			Location	Remarks
		Height	Width	Depth		
Central timing equipment	12	5.8	9.4	9.5	Internal (could be external if sheltered in appropriate environment)	
Up-data link	18.5	5.6	9.4	13	Internal (could be external if sheltered in appropriate environment)	
VHF multiplexer	12	5.8	12.3	16	Internal (could be external if sheltered in appropriate environment)	
TV camera equipment	4.5	8.25	3.5	16.5	Internal; portable; requires stowage and appropriate mounting brackets in laboratory	Definite need not extablished
Scimitar-notch antennas (2)	12.5 (each)				External part of CM outer structure	Will require new mounting provisions if crew compartment heat shield is not used for laboratory
SM rendezvous transponder	15	9	11	18	Internal or external with coldplate provisions; must be in proximity to transponder antennas	Required for rendezvous missions only
or						
Gemini rendezvous transponder with booster regulators	32.5 5	21.4 8.1	10.1 4.2	9.9 7.1	Internal or external with coldplate provisions; must be in proximity to transponder antennas	

Table 26. Independent RCML, Block I, Anticipated Communications/Data System (Cont)

Equipment	Weight (lb)	Dimensions (in.)			Location	Remarks
		Height	Width	Depth		
SM rendezvous antenna with waveguide	30 (approx)				External; close to transponder; appropriate aperture pointing angle and environmental protection required	Required for rendezvous missions only
or						
Gemini rendezvous spiral antennas (2) and dipole antenna	20 (each) 1.5	8.1 dia x 4.2 6.5 dia x 10.7				

INDEPENDENT RCML - BLOCK II - EQUIPMENT AND RENOVATION REQUIREMENTS

Table 27 summarizes the renovation requirements and Table 28 provides the equipment complement and physical characteristics for an independent RCML-Block II. The equipment is identical to the Block II vehicle communications/data subsystem with the following exceptions: For the same reasons as in the Block I laboratory, there is no requirement for HF and VHF recovery equipment, C-band transponder, and the FQR. There is a requirement, applicable to rendezvous missions only, for a rendezvous transponder and associated antennas. New mounting provisions must be designed for the RR/T, VHF scimitar, and S-band omniantennas. The RR/T must be provided with coldplate cooling and its antenna must be mounted in proximity to the electronics enclosure to minimize RF losses.

The VHF scimitar antennas are mounted on the service module for the Block II vehicles. Since they are not recoverable items, new antennas as well as new mounting provisions will be required.

In all probability, the heat shield will not be used for RCML models. Consequently, an alternative means for mounting the S-band omniantennas must be provided. The replacing shroud or structure should provide a ground plane and mounting provisions equivalent to those provided by the aft heat shield.

EXPLORATORY AND SPECIAL TESTS

Exploratory testing for communications/data equipment is limited to analysis or diagnosis required to determine the general nature and the approximate degree of degradation. Visual inspection, go/no-go tests, and various selected parameter tests will constitute the array of exploratory tests for communications/data equipment.

Figure 50 illustrates a flow chart depicting the testing for RCML communications/data equipment. Table 29 gives a preliminary estimate of the types of selected parameter tests that will have to be performed. Supplier renovation certification would be required for the RCMS communications/data equipment. In both cases, however, acceptance-type test will be the means for requalifying all equipment.

No firm need can be identified at this time for special tests on communications/data equipment. However, the following tests might be advisable.

Table 27. Block II Laboratory Renovation Requirements, Communications/Data System

Equipment	Test and Reuse	Replace	Not Required	Refurbish Cost	Refurbish Description	Remarks
Data storage equipment				~$5000 for typical overhaul	Diagnostic preacceptance test; possible replacement of items such as bearings, belts, idlers, springs, seals, motors, and pressure valve; alignments; acceptance-type tests	Remaining DSE life in hours will be assessed by supplier; 30-day turnaround time.
S-band omniantennas (4)				~$1700 (each) 100 percent of original operations cost	1. Removal from heat shield 2. Cleaning 3. Antenna inspection and testing 4. Modification of antenna mount for thinner heat shield, if required 5. Reinstallation of antennas in refurbished heat shield 6. Test of reinstalled antennas	1. Special tools and methods required 2. Special procedure and possibly tools required. 3. No comment 4. Design effort might be required 5. No comment 6. No comment
S-band power amplifier				Cost of TWT plus the same cost factor as in "Test and Reuse"	Replacement of TWT traveling wave tube	TWT supplied by Hughes Aircraft Company
TV camera equipment				Cost of vidicon plus the same cost factor as in "Test and Reuse"	Replacement of vidicon	TV camera supplied by Westinghouse
VHF antenna release mechanism (2)			X			
HF transceiver			X			
VHF/AM transceiver	X					
VHF recovery beacon			X			
Unified S-band equipment	X					
Audio center	X					
Premodulation processor	X					
VHF triplexer	X					
Signal conditioning equipment	X					

Table 27. Block II Laboratory Renovation Requirements, Communications/Data System (Cont)

Equipment	Test and Reuse	Replace	Not Required	Refurbish		Remarks
				Cost	Description	
PCM telemetry	X					
Central timing	X					
Up-data link	X					
S-band antenna switch	X					
VHF antenna switch	X					
RF and video cables	X					
HF recovery antenna			X			
VHF recovery antenna (2)			X			
C-band transponder			X			Remove and place in storage
C-band antennas (4)			X		Remove from heat shield; plug holes in heat shield	Special tools and procedures required for removal
Flight qual recorder			X			Remove and place in storage

Table 28. Independent RCML, Block II, Anticipated Communications/Data System

Equipment	Weight (lb)	Dimensions (in.) Height	Width	Depth	Location	Remarks
Unified S-band equipment	38	6	9.5	21	Internal (could be external if sheltered in appropriate environment)	
S-band power amplifier equipment	30	6	5.75	22.25	Internal (could be external if sheltered in appropriate environment)	
VHF/AM transceiver No. 1	11	6	4.7	12	Internal (could be external if sheltered in appropriate environment)	
VHF/AM transceiver No. 2	11	6	4.7	12	Internal (could be external if sheltered in appropriate environment)	Required in EVA missions only
VHF triplexer	2	3.4	4.6	5	Internal (could be external if sheltered in appropriate environment)	
Audio center	7	6	4.7	8.7	Internal (could be external if sheltered in appropriate environment)	
Premodulation processor	14.5	6	4.7	10.6	Internal (could be external if sheltered in appropriate environment)	
Signal conditioning equipment	45	6	9.8	20.25	Internal (could be external if sheltered in appropriate environment)	
PCM telemetry equipment	44	13.3	7.75	14.2	Internal (could be external if sheltered in appropriate environment)	
Central timing equipment	10	6	4.7	8	Internal (could be external if sheltered in appropriate environment)	
Up-data link	19	5.6	8.9	17.75	Internal (could be external if sheltered in appropriate environment)	
Data storage equipment	40	6	22	9.5	Internal (could be external if sheltered in appropriate environment)	Definite need not established
Television camera equipment	8.8	6.5	3.6	18 (approx)	Internal; portable; requires shock-proof stowage and mounting location for picture taking	Definite need not established
SM rendezvous transponder or	15	9	11	18	Internal or external with coldplate provisions; in proximity to rendezvous antennas	Rendezvous missions only
Gemini rendezvous transponder with booster regulations	32.5	21.4	10.1	9.9		
	5	8.1	4.2	7.1		

Table 28. Independent RCML, Block II, Anticipated Communications/Data System (Cont)

Equipment	Weight (lb)	Dimensions (in.) Height	Width	Depth	Location	Remarks
SM rendezvous antenna or	3 (approx)				External; in proximity to transponder with protective radom or equivalent; requires appropriate pointing	Rendezvous missions only
Gemini rendezvous spiral antennas (2) and dipole antenna	2 (each)	8.1 dia x 4.2 (each)				
	1.5	6.5 dia x 10.7				
S-band omniantennas (4)	2.5 (each)				External on CM structure (same locations as in CM spacecraft); require new mounting provisions and protective radoms or equivalent	
VHF omniantennas (scimitar) (2)	12.5 (each)				External; require appropriate pointing; new mounting provisions, and protective cover on the cables side	
S-band high-gain antenna	45				External on deployable boom	Not required in low-altitude earth-orbit missions
High-gain antenna electronics assembly	15.5	12.9	4.6	15.5	External; in proximity to S-band high-gain antenna	

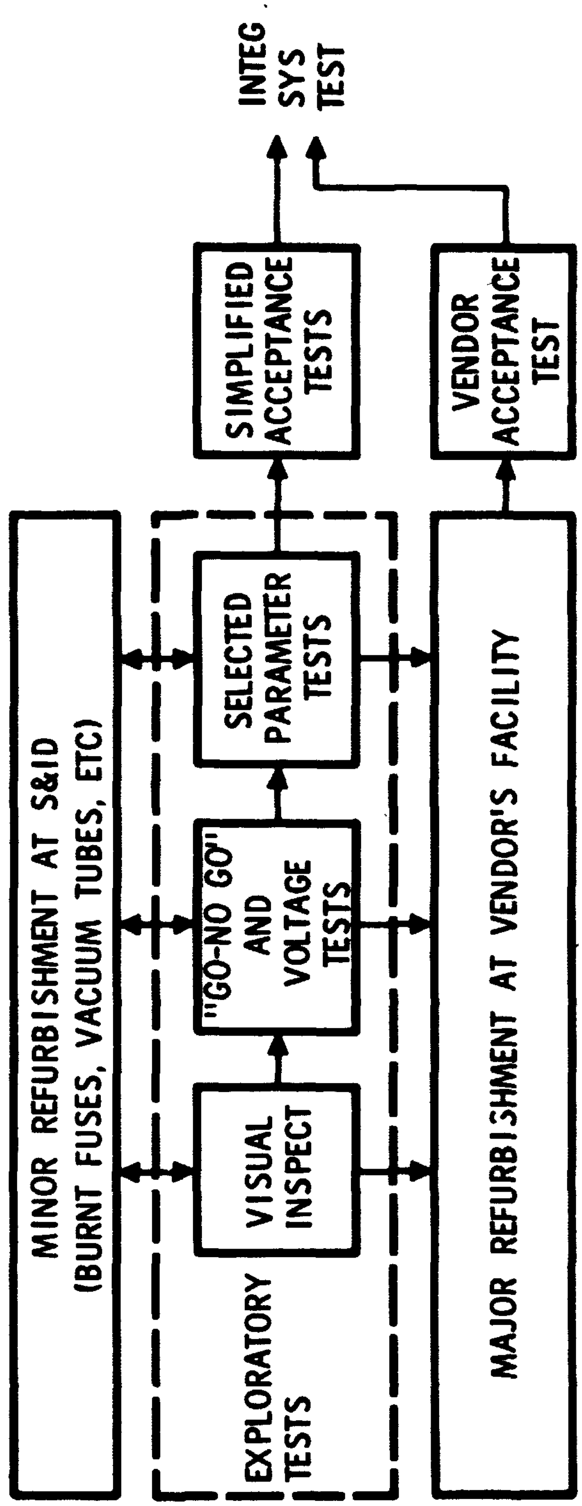

Figure 50. Flow Chart of Exploratory Tests

Table 29. Communications/Data Exploratory Tests (Preliminary)

Equipment	Vehicle			Types of Tests
	RCMS	Block I Laboratory	Block II Laboratory	
Data Storage Equipment	X	X	X	Verify the capability to record and playback voice and PCM telemetry signals
S-band omniantennas (4)	X	-	X	1. Subject to cleaning operation to remove char and RTV residue from antenna 2. Subject to VSWR tests
S-band power amplified	X	X	X	Power output
TV camera equipment	X	X	X	Scan test in limited light area
HF transceiver	X			1. Verify voice transmission and reception in SSB and AM mode 2. Check power output and carrier frequency
VHF/AM transceiver	X	X	X	1. Verify voice transmission and reception 2. Check power output and carrier frequency
VHF recovery beacon	X	-	-	1. Power, frequency, and modulation outputs
Unified S-band equipment	X	X	X	1. Verify capability to transmit and receive analog and digital signals 2. Check power output and carrier frequency
Audio center	X	X	X	1. Verify voice intercom capability and control modes
Premodulation processor	X	X	X	Verify modulation, demo .lation, mixing, and switching capabilities - in conjunction with testing of the interfacing communication/data units
VHF triplexer (multiplexer)	X	X	X	1. Check insertion loss, VSWR, and cavity tuning

Table 29. Communications/Data Exploratory Tests (Preliminary) (Cont)

Equipment	Vehicle			Types of Tests
	RCMS	Block I Laboratory	Block II Laboratory	
Signal conditioning equipment	X	X	X	1. Verify signal parameter conversion
PCM telemetry Block I		X		1. Verify presence and configuration of sync pulses
PCM telemetry No. 2, Block I		X		2. Verify response to various d-c input levels
PCM telemetry, Block II	X		X	
Central timing equipment	X	X	X	1. Check amplitude and frequency of all timing outputs
Up-data link	X	X	X	1. Verify test message 2. Validity code check
HF recovery antenna	X	-	-	Visual inspection
VHF recovery beacon antenna (2)	X	-	-	

1. In the RCML, the heat shield probably will not be used and will be replaced by some type of shroud. The shroud shape and the characteristics of the material used for the shroud in the immediate vicinity of the S-band omniantennas of Block II and the SCIN antennas of Block I will affect the radiation patterns of the antennas. Similar considerations apply to the rendezvous and VHF scimitar antennas which in Block II are located on the SM. In the RCM lab configuration, these antennas will probably be mounted on the cruciform structure. Thus, antenna radiation pattern tests might be required. The purpose of the tests would be to ascertain the correctness of RCML design modifications. The tests, therefore, would be non-recurring and could probably be conducted on scaled-down models.

2. The output of the PCM telemetry contains information on the characteristics and status of various CM subsystems and components. This suggests the possibility of using the PCM telemetry as a test equipment to obtain rough diagnostic data on the subsystems degradation and renovation requirements. This test would have to be limited to the CM components which would not be damaged by the application of power. Means for supplying power to the subsystems and components under test would have to provided.

SYSTEM FUNCTIONAL CAPABILITIES

Communication/data capabilities vary only slightly among the three versions of the RCM vehicles. The functional capabilities of the RCMS can be summarized as follows.

The basic functions performed by the communications/data subsystem are: time, voice, command, television, telemetry, tracking and ranging, recovery and scientific data storage, and transmission.

Timing function and signals are provided by the central timing equipment and are used to synchronize various data trains within the spacecraft subsystem equipment.

Voice function is provided by the audio center working in conjunction with the VHF equipment, HF transceiver, premodulation processor and S-band equipment.

Command function is provided by the up-data link operating in conjunction with the premodulation processor and the S-band equipment. This function permits updating the central timing equipment and the guidance and navigation equipment, as well as furnishing real-time commands to the spacecraft subsystems.

Television coverage of spacecraft activity can be implemented by use of the TV camera in conjunction with the wide-band modulation provisions of the S-band equipment and premodulation processor.

Telemetry capability is provided by the PCM equipment working in conjunction with the signal conditioning equipment, the premodulation processor, and the S-band equipment.

Tracking and ranging functions are provided by the coherent transponding capabilities of the S-band equipment.

Recovery aids are provided by both the HF transceiver and the VHF recovery beacon. The HF transceiver is capable of single side-band voice operation, amplitude-modulation voice operation, and CW beacon operation.

Scientific data transmission is accomplished through the use of the premodulation processor and S-band equipment. Scientific data acquisition and instrumentation equipment are not provided.

The voice, PCM data, and scientific data can be recorded by the data storage equipment for subsequent playback and/or transmission through the premodulation processor and S-band equipment.

The functional capabilities of Block I and II independent RCML's are identical to those of the RCMS except that no recovery functions are provided in the laboratories, while the transponding capability for rendezvous missions is included.

VIII. DISPLAYS AND CONTROLS

The major environmental hazards to the display and controls (D&C) and instrumentation equipment will be vibration, high temperature, and high humidity. It is safe to assume that environmental degradation will be small unless the CM has been subjected to extreme levels of any of these parameters. Extreme levels may be taken to mean levels considerably higher than the design-mission levels already experienced or anticipated. Malfunctions in the ECS which permit high humidity for extended periods will require careful assessment of components for reuse. An analysis by environmental hazards leads to the following conclusions:

1. Hard vacuum should affect no components lastingly, unless it is maintained for excessive periods.

2. Radiation will tend to degrade electroluminescent material. It will also temporarily degrade the operation of equipment using semiconductor devices. A radiation level sufficiently high to cause permanent degradation of semiconductors would provide a personnel hazard.

3. Meteorites will result in damage only through direct hits on the main display console (MDC). (This is highly improbable but, most likely, would be catastrophic.)

4. High temperatures will cause potential or eventual degradation of nearly all types of components and definitely reduced reliability. Temperatures must go considerably higher than anticipated, e.g., coldplate failure, in order to cause degradation.

5. Low temperatures will have little effect unless they are accompanied by vibration or impact.

6. Vibration may affect meters (as noted), circuit breakers, incandescent lamps, connectors, and floodlights. Extreme levels of vibration will render all of these items suspect.

7. Impact (high levels) will affect the same items affected by vibration, and the same caution applies.

8. Condensation will affect certain low-current, AT-type switches, circuit breakers, and connectors which, if not hermetically sealed, may be subject to moisture invasion with attendant dielectric changes.

It is rather difficult to assess operational degradation of MDC items on the basis of complete panel assemblies. Since similar items appear on different panel assemblies in varying numbers, it is preferable to analyze potential degradation by components and to extrapolate refurbishment requirements from anticipated degradation.

Switches appear to be one of the main problem areas with regard to present life expectancy. Rotary switches have the highest reliability and will most likely be good for reuse in extended missions without refurbishment. Bush-button switches, although present on several panels, are rarely used during a mission and then only when a manual backup function is required. One exception to this criterion is the Master Alarm (MA) reset switch, which will be operated from time to time during the mission. This switch will be covered separately under Caution/Warning (C/W) System. Based upon the small number of operations, the performance degradation of the push-button switches may be assumed to be not highly significant.

Toggle switches present an entirely different situation. Their operational life expectancy is presently in doubt and considerable difficulty has been experienced in obtaining suitable toggle switches for the Apollo mission environment.

Design problems have included internal actuator contamination, actuator breakage, melting, and contact shorting among others. Life testing has not been satisfactorily performed on present switches, although it must be anticipated that by the time of Block II flight-article production, a satisfactory supply of toggle switches with a life test of several mission periods will be available.

A major problem in estimating the degree of potential degradation is the lack of information regarding the number of operating cycles on any particular switch during preflight and mission operations.

Circuit breakers of Block II apparently pose no problems for reuse. The only exception to this would be wherein the circuit breaker had been abnormally operated manually as a switch a number of times. Each manual pull of the circuit breaker slightly lowers the current required to trip the device and may result in premature operation of the device.

Vertical meters appear to be satisfactory for reuse, because the only potential operational degradation will be an increase in frictional error due to the tight-pivot adjustment required for resistance to vibrational effects.

Round meters are inherently more rugged and need to meet less stringent accuracy required (5 percent) than do vertical meters (2 percent). The round meters should be available for reuse, after undergoing only integrated-system checkout.

Connectors may pose a problem because of potential damage inflicted during the removal and installation of panels.

Event indicators may be broken down into electromechanical and incandescent ty •. Electromechanical indicators should not be significantly degraded but p. .ently are experiencing some problems in operation, e. g., both failure to respond and failure to return have been noted.

Incandescent lamps follow a general pattern of predictable limited life. Especially significant is the correlation between color, temperature, brightness, and life expectancy, which are functions of operating voltage versus design voltage. In order to obtain a sufficiently bright warning signal, long life must be sacrificed to some degree.

Lighting may be broken down into three categories: integral, flood, and exterior.

Integral lighting consists of electroluminescent (EL) panels bolted to the main display console and provides low-level illumination of the panel for switch legends, meters, controls, etc. Brightness controls are supplied to permit maintenance of dark adaption for crewmen (Block II only).

Electroluminescent lighting in use suffers from continual degradation of total light output. This is believed to be a result of crystal structure changes through continual electron bombardment and the presence of trapped water in the plastic. The amount of degradation is a function of voltage and frequency of the electrical supply and is accelerated by heating and radiation. The light output degradation follows approximately an exponential curve, the major diminution of light occurring during the period of preflight and acceptance testing. See Figure 51. From the brightness at time of manufacture, there will be a reduction of about 45 percent by the end of 150 hours of operation. (Even greater reduction will result if operated at elevated temperatures.)

Based upon the light output at the start of a 14-day mission, there will be a 35-percent reduction in light at the end of the 14-day mission.

- **LIGHT OUTPUT**
  DEGRADES WITH OPERATING TIME

- **INCREASING VOLTAGE**
  PERMITS CONSTANT LIGHT OUTPUT LEVEL WITH OPERATING TIME

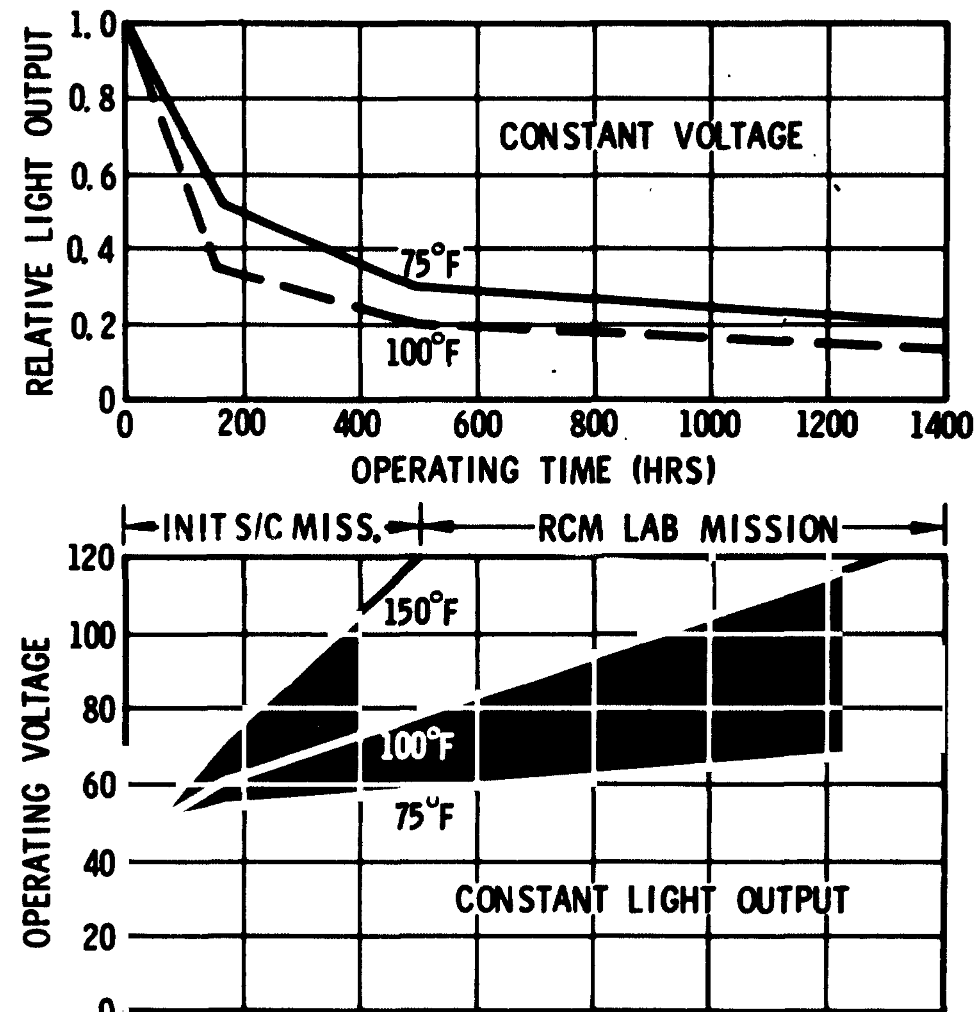

Figure 51. Electroluminescent Lamps

If the EL lighting is to be used for a subsequent 30-day mission (plus 150 hours at checkout), there will be only a 30-percent reduction from start to finish of this extended mission. Assuming that light output is adequate at the finish of the original 14-day mission, it should be close to the same value at the end of 30 days.

Another consideration in EL life expectancy is the concept of constant-brightness illumination (Figure 51). Considerably longer life is experienced when the panel lamps are turned to a suitable brightness initially and then maintaining that light level by periodically increasing the input voltage with variable controls.

Floodlighting is accomplished by fluorescent lighting which is not of the household type. About half of the lamps contain filaments which indicates that the floodlighting should follow the same restrictions placed on incandescent lamps. Floodlight power supplies should suffer very little degradation.

Postlanding beacon lighting is designed for postlanding impact operation and has a very high probability of survival for reuse. Since the beacon is exposed, however, the possibility of salt-water immersion or damage at time of retrieval is quite high.

Variable controls consist of rheostats used for lamp dimming, rotary (variable) transformers used for the same purpose, and potentiometers. Potentiometers are used in communications volume controls, squelch circuits, and for certain functions in the ECS. The probability of operational degradation of any of the variable controls appears rather low and their reuse seems entirely feasible.

The caution/Warning (C/W) system consists of a solid-state detection unit, an indicator matrix, and associated controls. Controls consist of toggle switches (as discussed previously with, all comments applying) and illuminated push-button switches. The primary difference between the Master-Alarm (MA) push button and the push-button switches previously discussed is the number of cycles of operation. One of three MA switches must be operated to reset the tone and lamp alarm each time any caution or warning is given, as well as each time a lamp test is performed on the C/W system.

The detection unit is a highly reliable unit not subject to operational degradation, with the exception of long-term drift of warning levels and of the possible degradation of internal power supplies due to continuous use and mean-time-between-failure (MTBF) considerations.

Other indicators include the clocks and timers, altimeter, and G-meter. All items in this group will experience very little operational degradation, with one exception: the mission and phase timers are GFE electronic digital-readout clocks with EL characters. The constraints regarding integral EL lighting also apply to the mission timers.

SUMMARY OF ANTICIPATED DEGRADATION

Those items most likely to experience degradation during operation are:

Electroluminescent lighting
Fluorescent lamps
Incandescent lamps
Electromechanical event indicators
Toggle switches
Master-Alarm (MA) reset switches
Caution Warning Detection Unit (CWDU) power supplies
Panel connectors

Of the above-listed items, all incandescent and fluorescent lamps, event indicators, and the CWDU power supplies will be definitely degraded and therefore are not candidates for reuse. Electroluminescent lighting will be conditionally suitable for reuse based upon light output level. Toggle switches, MA reset switches, and panel connectors will be degraded as a function of the number of operating cycles.

RENOVATION REQUIREMENTS

Since the number of operating cycles on any switch is not known and the toggle switches presently employed on the main display console (MDC) are marginal with respect to life-test requirements, it has been assumed that all toggle switches will be replaced. In lieu of mass replacement, a testing procedure is outlined in a following section.

MA reset switches will require relamping and testing at the switch. They may be replaced at the same time as are the toggle switches mentioned above.

Panel connectors should be replaced from a reliability standpoint. Since the connectors and approximately 50 percent of the panel connectors would be replaced, the panel wiring harness should be replaced at the same time. Table 30 indicates the number of each assembly component affected by the renovation activity.

Figure 52 indicates the location of these items.

Table 30. Assembly Components Affected by Renovation

Item No.	Replacement Need
BK II MDC PAN. 1 V36-761011	45 toggle switches Abort indicator Engine indicator MA switch NO-AUTO-ABORT switch Wire harness 9 connectors
BK II MDC PAN. 2 V36-761012	70 toggle switches 19 event indicators 2 C/W indicator matrices 9 connectors Wire harness
BK II MDC PAN. 3 V36-761013	62 toggle switches 21 event indicators 1 MA switch 10 connectors Wire harness
BK II MDC PAN. 4 V36-761014	3 toggle switches 2 connectors Wire harness
BK II MDC PAN. 7 V36-761017	3 toggle switches 4 connectors Wire harness
BK II MDC PAN. 5 V36-761015	3 toggle switches 5 connectors Wire harness
BK II MDC PAN. 8 V36-761018	10 toggle switches Wire harness 5 connectors
BK II MDC PAN. 6	5 toggle switches 2 connectors Wire harness
BK II MDC PAN. 9 V36-761019	8 toggle switches 3 connectors Wire harness
BK II MDC PAN. 10 V36-761020	5 toggle switches 2 connectors Wire harness
BK II MDC PAN. 11 X V36-762065	
BK II MDC PAN. 13 X V36-762013	
BK II MDC PAN. 98 X V36-764098	
BK II MDC PAN. 99 V36-764099	5 toggle switches 4 connectors Wire harness
BK II MDC PAN. 97 V36-764090	8 toggle switches 1 connector Wire harness

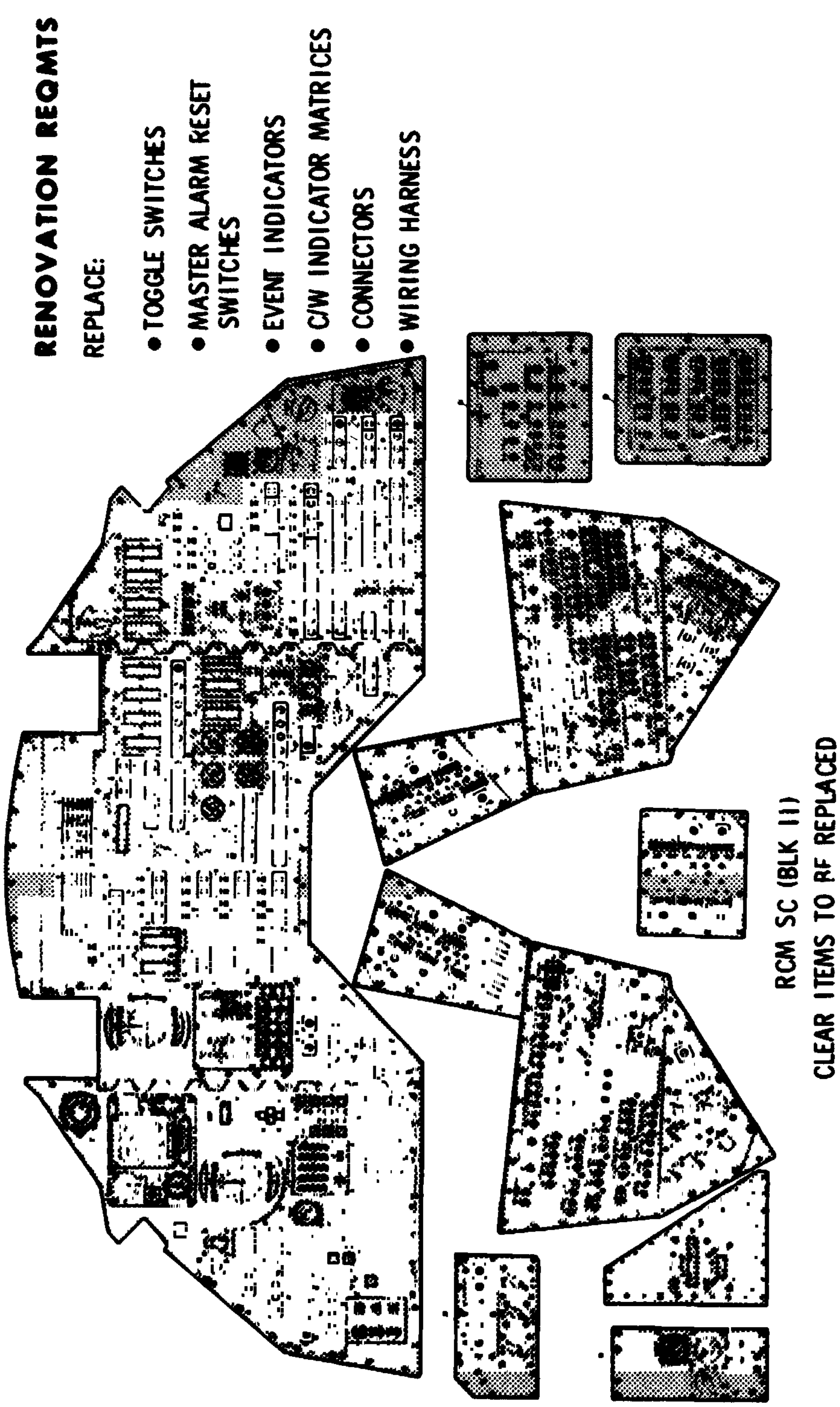

Figure 52. Main Display and Control Panel Renovation

POSTRECOVERY CONSTRAINTS AND REQUIREMENTS

Appropriate means and procedures are required for the desalinization and anticorrosion preservation of the recovered CM and its components. No other constraints or requirements for the postrecovery operations appear to be applicable from the standpoint of the displays/controls subsystem.

EXPLORATORY TEST REQUIREMENTS

Figure 53 indicates the sequence of events in manufacture, assembly, and testing of electronic subsystems in the Apollo CM. The right-hand time line indicates the normal sequence of a CM being built under initial procurement. The left-hand line indicates a tentative sequence for a recovered CM undergoing renovation. Circled numbers on the figure indicate items which may require clarification:

1. Assembly manufacture applies to such items as MDC panels, all components of which have completed acceptance testing at the various vendors from whom they are obtained. Assemblies are fabricated at Downey and are next tested by manufacturing. Items which are subcontracted assemblies do not undergo this nor the succeeding step.

2. This step is indicated in the event a system check is deemed necessary. The GSE umbilical, which will have been subjected to entry heat, must be removed and, preferably, replaced. The removal of this item is required to preclude electrical shorts in the various systems caused by a degraded umbilical.

   Power may be supplied to the command module through the refurbished GSE access connector, or the flight batteries may be removed and ordinary lead-acid batteries subsituted.

3. With either a battery or external power, most CM subsystems may be exercised and the MDC used to monitor performance. These tests may be used to provide information regarding the necessity for removal of subsystem assemblies for refurbishment.

   One specific test which may be made advantageously at this time is the check of EL integral-panel lighting. Because of the various peculiarities of the phosphorus required for white EL illumination, accurate photometric measurements are difficult, if not impossible. Color differences and brightness differences between various items on the MDC are obvious to the eye although not measurable by any present photometric equipment. This is because of the interrelation of color and brightness and the low light levels of EL panels.

The most satisfactory test probably will be a visual inspection by a trained technician having good visual acuity and color perception. Comparison of the various lighted elements at different light levels will provide an excellent basis for decisions whether to reuse or to replace. Extreme variation in color or brightness, scratches, spots, peeling, and flaking will be cause for rejection.

(4) Before it can be assumed that it is not necessary to replace all toggle switches and the MA reset switches, each switch must first undergo vibration testing and then be tested for contact resistance, insulation strength, and operating force. Failure of a switch on any of these tests will indicate necessary replacement of the item.

The CWDU will be removed at this time, its power supplies replaced, and a bench test performed. Bench testing will consist of a check of alarm limits, event outputs, MA triggering, and output signals.

(5) Refurbishment of MDC panels consists of replacement of EL lighting, where necessary, defective or marginal switches, and of all event indicators (incandescent or electromechanical). CWDU refurbishment will have been partially accomplished by replacement of power supplies. Further out-of-tolerance conditions will require replacement of affected modules.

(6) Two blocks, system test and combined system test, have been duplicated under Operational Checkout Procedures (OCP). This duplication is for emphasis on calibration checks of displays and instrumentation. No change is anticipated in OCP's. Frictional error in vertical meters may be determined at the time of integrated checkout by checking zero setting, full scale deflection, and return to a mid-scale reading.

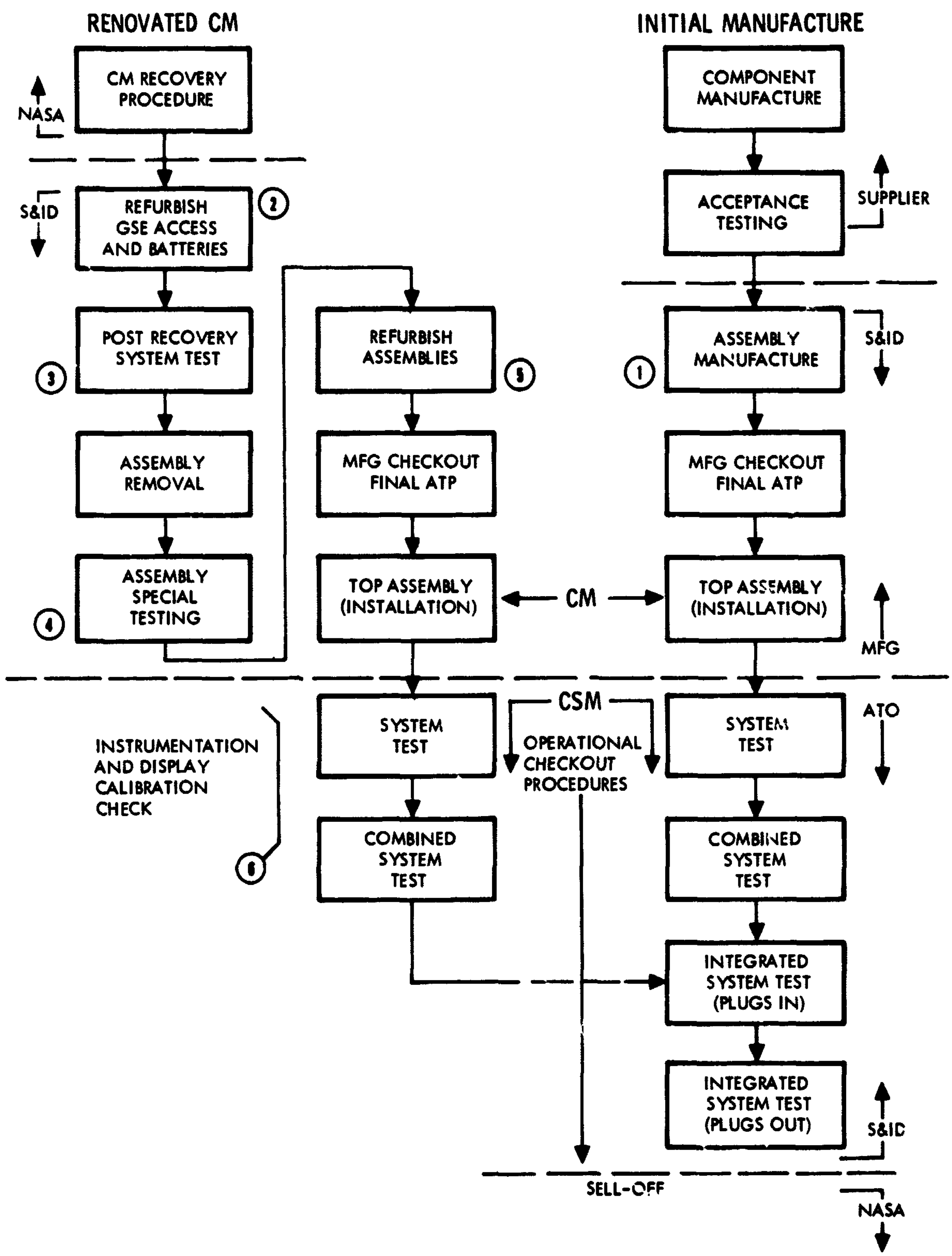

Figure 53. Test Requirements CM Assembly and Renovation

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IX. INSTRUMENTATION

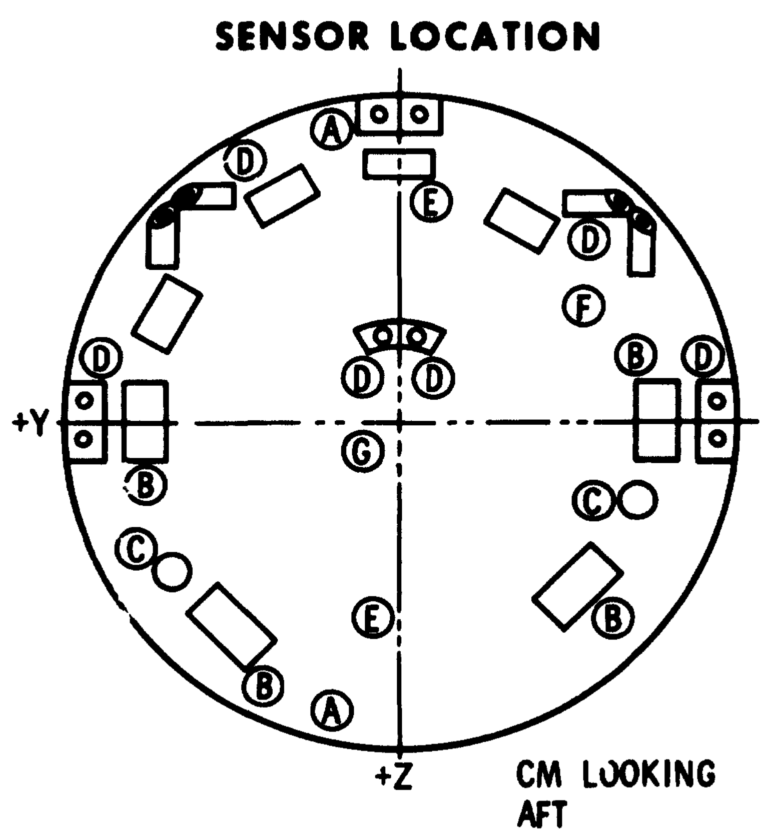
Operational instrumentation co sidered in this section covers the following measurements: (See Figure 54)

CA1800 T	
1803 T	Heat shield bond temperature
1806 T	
CA1809 T	
CC0188 P	Battery compartment manifold pressure
CF0006 P	Surge tank pressure
CR0001 P	
0002 P	H_e tank pressure and temperature
0003 T	
0004 T	
0005 P	Fuel tank pressure
0006 P	
0011 P	Oxidizer tank pressure
0012 P	
2100 T	
2103 T	
2114 T	Engine injector valve temperature
2116 T	
CR2119 T	
CS0220 T	Docking probe temperature
CS0100 X	CM-SM separation

All other items on the measurement list fall under one of the following cateogories:

1. R&D measurments: not required for operational instrumentation.

2. Subsystem instrumentation or measurements: integral part of subsystems.



TEST AND REUSE

- (A) COMPT PRESSURE
- (B) TANK PRESSURE (FU, HE, OX)
- (C) TANK TEMPERATURE (HE)
- (D) RCS ENG INJ TEMP (ORIGINAL VALVE ONLY - OTHERWISE REPLACE)

NOT REQUIRED

- (E) HS BOND TEMPERATURE

REPLACE

- (F) CM/SM SEPARATION
- (G) DOCKING PROBE TEMP

Figure 54. Instrumentation

3. SM, LEM, adapter, booster, and LES instrumentation; not applicable.

Of the twenty measurements to be considered, Numbers CA1800-1809 are heat-shield bond-temperature sensors. They are located on the inside of the aft heat shield and may suffer degradation due to entry heating, although this is unlikely. These measurements are similar to flight-qualification measurements and may be deleted in the renovated command module.

CC0188 battery compartment manifold pressure instrumentation will be reusable and will suffer no degradation in operation.

CF0006, CR0001, 2, 5, 6, 11, and 12 instrumentation will suffer no operational degradation and may be expected to be reusable. Calibration may be rechecked during system pressurization . CR0003, and 4 will experience no degradation and will be reusable.

CR2100, 3, 10, 14, 16, and 19 are temperature sensors that will undergo no deterioration and may be reused. These temperature sensors are bonded directly to the RCS engine-injector valves and may be reused only if the same RCS valves are reused. Any attempt to remove the devices will most likely result in an unusable device. This restriction applies to all bonded temperature sensors.

Since instrumentation devices are sealed and potted, salt-water immersion may lead to some apparent degradation in the external appearance of the equipment even though its operation will be unimpaired.

Instrumentation equipment, in general, appears to be reusable in all cases, when used in conjunction with the original CM equipment. Replacement of instrumented equipment employing bonded sensors will generally require replacement of instrumentation. Table 31 shows the instrumentation renovation requirements.

Table 31. Instrumentation Renovation Requirements

Item	Test and Reuse	Replace	Not Req.	Remarks
V36-757525 4 ea 757522			X	HS bond temp
V36-757521 1 ea	X			Batt comp press
V36-753521 1 ea	X			Surge chamber press*
V36-757527 757522 6 ea	X			Tank press H_e, fuel, oxid*
V36-757526 757522 2 ea	X			Tank temp H_e*
V36-757533 757523 6 ea		X		Eng inj heat temp***
F01-750009 1 ea		X		CMSM Sep monitor
V36-751521 1 ea		X		Docking probe temp
V36-751870 1 ea	X			Radiation survey meter (GFE) RFB OP42-001 Formerly Mfg. by EON

*With original tank
***May be reused with orig valve

X. LANDING, RECOVERY, AND DOCKING SYSTEMS

SYSTEM DEFINITION

For the purposes of this study, the earth landing system consists of the crew couches, couch struts, and the uprighting subsystem (Figure 55). The three crew couches are designed as a single unit supported by a series of energy-absorbing struts. The energy of landing is dissipated primarily by crushing honeycomb in a cylinder and, secondarily, by friction of pads sliding on the cylinder. The uprighting system is designed to prevent the command module from floating in an inverted position after water landing, thus providing a more comfortable crew-seating position and easier egress through the docking hatch. The system consists of three air bags initially stowed on the parachute deck and inflated at crew command.

The recovery system (Figure 56) is composed of mortar-deployed drogue parachutes and Ringsail parachutes with mortar-deployed pilot parachutes. All the parachutes are attached to the command module at a single fitting which is equipped with pyrotechnic cable cutters to release the parachutes. The mains are released at the discretion of a crewman after landing. Parachute-deployment sequence of events is automatically actuated by signals from the sequence controller and the pyrotechnic continuity verification box (covered in ection VI) which are located inside the command module. The sequence controller is capable of limited manual override by a crewman during the final minutes of flight.

The command module portion of the docking system consists of a probe mechanism which has the capability of mitigating shock loads experienced during the docking procedure. The mechanism is attached to the docking tunnel structure by an adapter ring equipped with a circumferential pyrotechnic cutting device for operating the mechanism from the command module (Figure 57). Built on the inside of this ring are four automatic latches used to draw the LM and CSM together to an initial position, and 12 manual latches for final positioning.

SYSTEM PERFORMANCE REQUIREMENTS

The performance of these systems cannot be degraded because of refurbishment and reuse of components. Many of these components are considered crew-safety items with reliability apportioned on this basis. These systems are peculiar, however, in that many system components

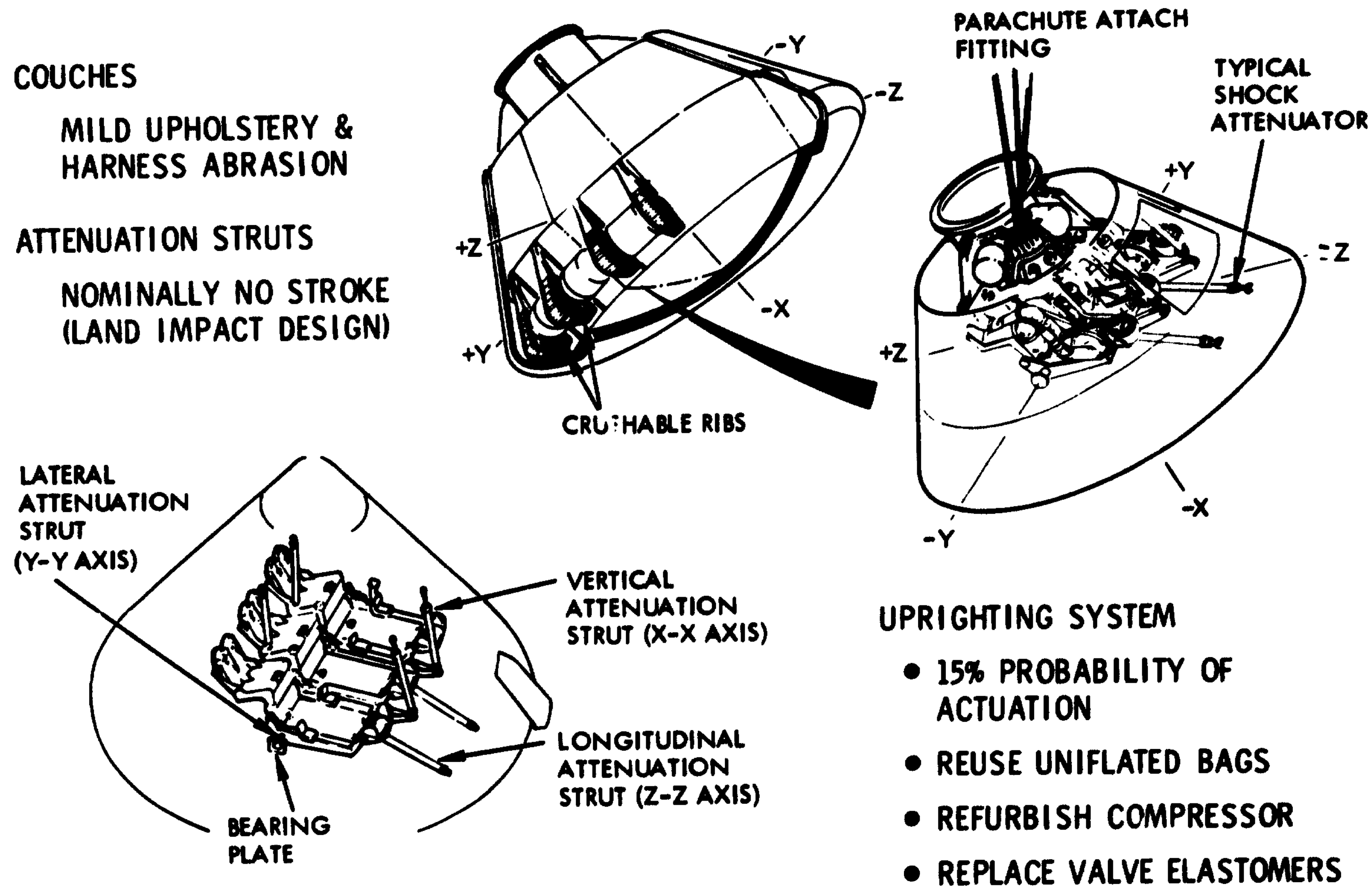
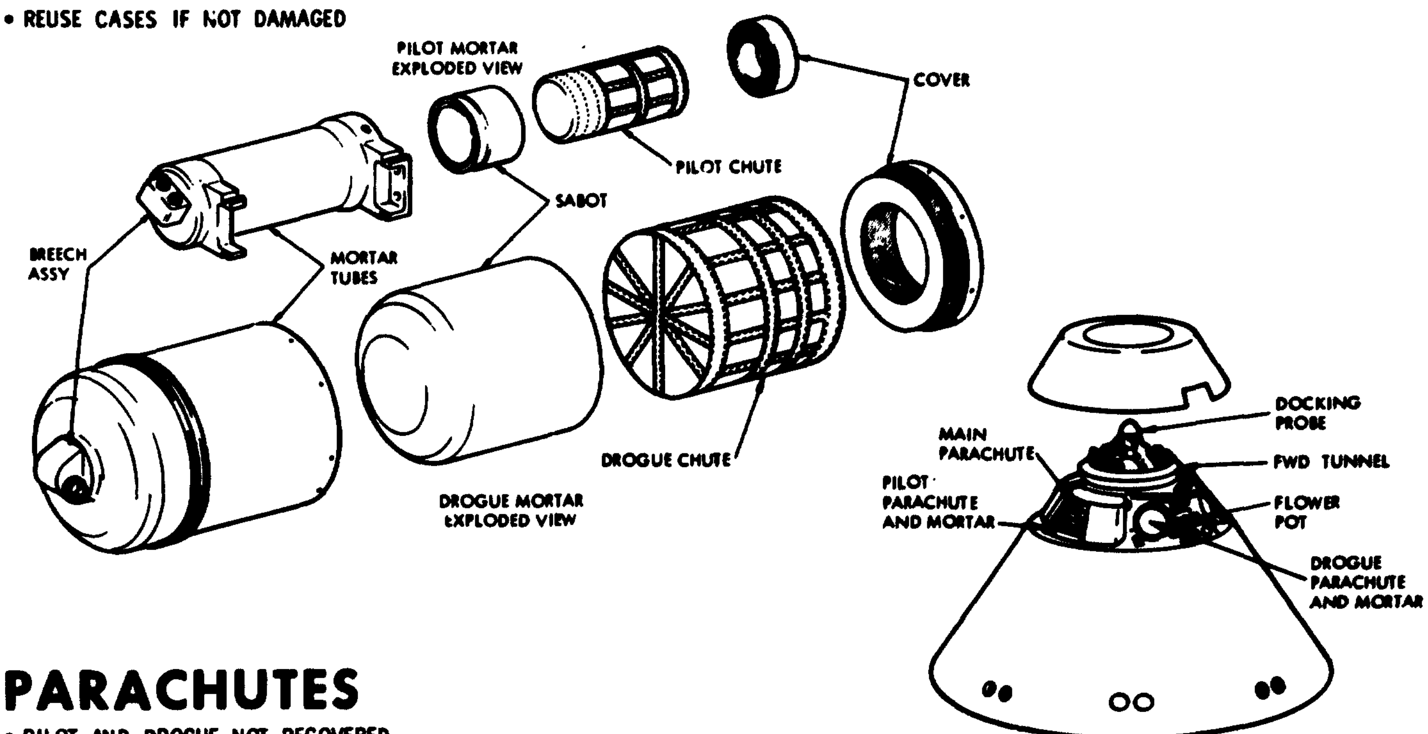


Figure 55. Earth Landing System

MORTARS

- SABOTS AND COVERS NOT RECOVERED
- REUSE CASES IF NOT DAMAGED



PARACHUTES

- PILOT AND DROGUE NOT RECOVERED
- MAIN MAY BE JETTISONED

PARACHUTE ATTACHMENT FITTING (FLOWER POT)

- RISER SCUFFING
- PYROS AND RISER CUTTERS ACTUATED

Figure 56. Recovery System

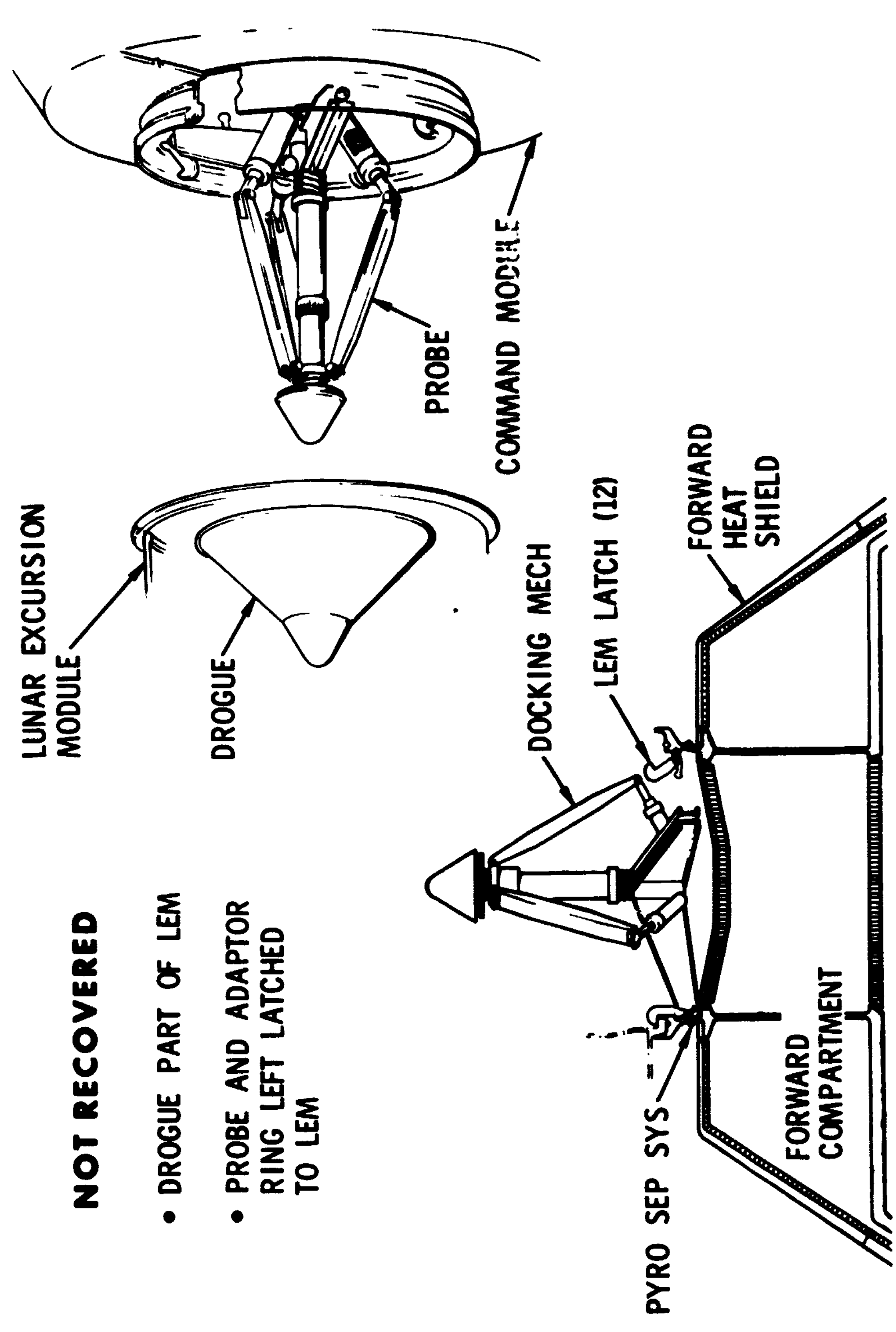


Figure 57. Docking System

are lost after their function is completed and must be replaced by new components, thereby providing better reliability than if components were reused. Materials and designs which are used in the recovery and earth-landing system were selected to minimize the degree of degradation of the components to ensure operating reliability after exposure to the space environment. This fact makes the components which are recovered candidates for refurbishment.

SYSTEM RENOVATION

The following pages discuss system-refurbishment capability in terms of degree of degradation for each system component after exposure to the environment, and the requirements for replacing components which are lost during normal operation.

Docking Mechanism

The docking mechanism falls into the lost-during-operation category. In the docked-orbital configuration, the docking probe is stowed within the command module, but prior to vehicle separation the probe is replaced in its original position. After hatch closure and vehicle separation, the adapter ring is separated from the docking hatch structure by the pyrotechnic cutter and the docking mechanism is not recovered. The renovated command module, therefore, must be equipped with a new docking mechanism.

Recovery System

During normal recovery-system operation, the drogue parachutes are severed from the command module at high altitude and are not available for reuse. At main parachute deployment, the riser of the pilot parachute is used as a ripcord to release the parachute from the retention bag. This fact alone precludes the reuse of the bag on further missions since the fabric generally is damaged. Even if damage were not sustained, removal, cleaning, drying, and inspection would nullify the benefit of reuse on this inexpensive item.

The main parachutes may be disconnected from the command module at the discretion of the crew. If this has happened, there is a chance that they can be recovered. The cost of recovery in this situation, however, is quite high relative to the cost of the parachutes, even when the parachutes can be located. Assuming that the parachutes remain attached to the command module, they will be saturated with sea water and probably will be colored by the dye marker. Cleaning and drying requires specialized equipment which is not readily available. Furthermore, the factor of safety of the parachutes has been decreasing as the command module weight has been increasing and this will result in increased minor damage to the

textile materials. Although this damage is of little consequence to the performance of the parachute, repair would be very difficult. Parachute reuse has its application in cargo drops where parachute safety factors are high and where occasional malfunction can be tolerated. But refurbishment and reuse is not recommended in Apollo manned missions for these relatively inexpensive components.

The two drogue parachute mortars are candidates for refurbishment, providing damage has not been sustained during parachute deployment. It has been found after many recovery system tests and test flights of hardware that the mortar cases are subject to damage from the parachute riser (Figure 58). For instance, on spacecraft 011, the thin aluminum shell in the muzzle area is bent and torn, making it totally unsuitable for refurbishment. When no damage is sustained from this cause, the mortars may be refurbished to the original condition. Degradation due to space environmental conditions will be minimal because the mortar is basically a metal assembly. Launch and entry heating have no affect on the mortar case. To refurbish the mortar, it will be necessary to remove the unit from the command module, strip the unit, and thoroughly clean away salt and pyrotechnic deposits. A new sabot and end-closure cap will be required in addition to replacement of all elastomers in the in the unit. The pilot parachute mortar may be refurbished in a manner similar to that defined for the drogue mortars, providing damage has not been sustained during operation.

The parachutes are attached to the command module by individual riser lines at the parachute attachment fitting, which is bolted to the CM structure on the Z-axis of the forward equipment compartment. This piece of hardware contains five cartridge-operated cutters with replaceable anvils for separating the parachute riser lines from the command module. The unit will degrade minimally in the environments to which it is subjected. Vacuum meteoroid and temperature have no effect on the fitting, but prolonged exposure to sea water may produce some corrosion in the area of dissimilar metal contact. Degradation due to operational use has yet to be determined, since the Block II has not yet been flight-tested. Scuffing of the fitting surface by the risers, however, is to be expected during the parachute deployment sequence. Refurbishment consists of removing the unit from the command module, disassembling, and cleaning. Scuff marks may be removed by grinding. Cutters and anvils should be replaced and elastomers renewed.

The drogue parachute risers are cut during normal operation and must, therefore, be replaced. On the other hand, the main parachute risers, are cut after landing, at crew option. It is recommended that these risers, even if they are intact after recovery, be discarded. The difficulty of cleaning out salt deposits in the cable and cable fittings, which may cause subsequent corrosion, is the reason for this recommendation.

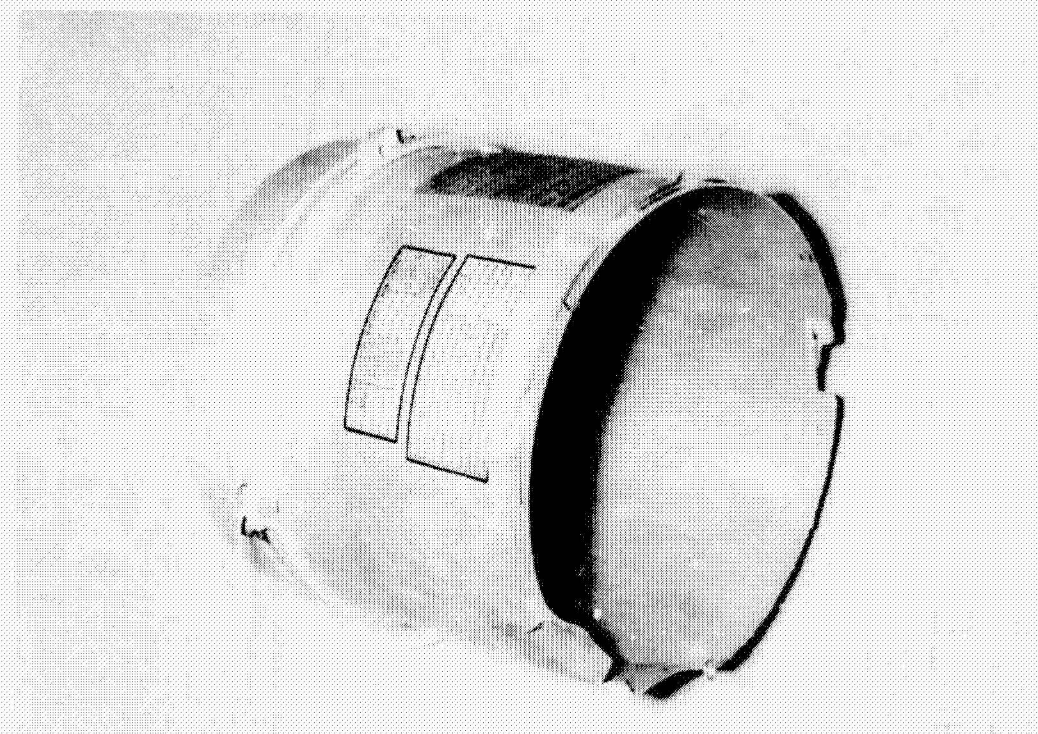


Figure 58. Spacecraft 011 Drogue Parachute Cannister Damage

The earth-landing sequence controller is located in the lower equipment bay. In this position, its degradation due to environment is at a minimum and, in fact, the controller is designed to function even after exposure to these environments. The functioning components consist of relays, solid-state timers, and switches operated by barometric pressure. Since the controller is functioned many times during checkout, it is recommended that all relays be replaced during refurbishment. However, as concluded in Section VI, the entire unit is best replaced due to its construction. The controller senses external atmospheric pressure at the command module side wall through a 3/8-inch diameter tube. After landing, this tube could be under water for short periods of time and should be cleaned before reuse. Salt-water intrusion into the sequence controller will be cause to scrap the unit. The possibility of this happening is remote but it could occur if sufficient water penetrated into the interior of the command module due to a malfunction in the pressure-equalizing valves.

The pyrotechnic continuity verification base, located adjacent to the sequence controller, is an integral part of the recovery system control. The life of this unit is limited by the number of operational sequences or cycles to which the unit is subjected during checkout, and the same rationale applied to the controllers with a replacement conclusion applies here.

Earth Landing System

The upright system consists of three air bags stowed beneath the parachute and inflated to a low pressure by air from a reciprocating compressor. Degradation of the bags from exposure to vacuum has been defined in tests in the vacuum chamber. Test results showed little change in properties due to a 14-day exposure. Temperature extremes in the stowage area will degrade the bag material only minimally. In operational use, the inflated bags will be subjected to the action of the sea causing a general abrasion of the bag material, particularly in the area of bag attachments. For this reason, activated bags which have been used are not recommended for reuse. The bag-pressurizing system consists of the compressor and check and relief valves coupled together with aluminum tubing. In operation, air is drawn from inside the command module and pumped into the bags. Degradation of the system from external environment during flight will be negligible. Operational use of the compressor, however, will gradually deteriorate the unit and internal corrosion due to pumping salt water is also a possibility. The operating life of the compressor has been set at three hours, but time consumed during checkout of the compressor and the uprighting system generally amounts to a considerable portion of this life. Refurbishment consists of stripping down the unit, cleaning, and inspection. The unit must be rebuilt with new elastomers and new valves.

The solenoid valve, pressure relief valve, and check valve may be reused after cleaning, inspection, and replacement of elastomers. New tubing should be installed to eliminate the possibility of using plumbing in which salt-water corrosion has started.

It has been estimated that the uprighting system will be deployed in only 15 percent of landings, relieving the requirement for major refurbishment.

As the first Block II crew couches have yet to be qualified, any analysis of refurbishment capability will be limited. Since the couches are designed for the land-landing case, however, low structural loading associated with water landing will present no obstacle to refurbishment. In fact, tests have shown that the couch attenuation struts will not stroke during a nominal water landing. Degradation of couch and struts due to environment is minimized by their location in the command module and by the design philosophy which requires the couch system to function after exposure to the environment. The degree of structural degradation is, therefore, expected to be negligible. On the other hand, synthetic organic materials will suffer degradation mainly from operational use. Visible wear, dirt, etc., will be cause to replace the crew restraint harness and couch covering. These components could not be reused if they have become saturated with water.

The struts must be removed and disassembled and, in the event that the strut show no stroking, the unit may be rebuilt and recalibrated. Struts which have stroked may be rebuilt and calibrated, using new crushable material.

The components which are recovered must be stripped down for inspection and cleaning during spacecraft refurbishment. Therefore, no special postrecovery operational constraints are required. Hosing down the parachute deck or forward equipment bay with plain water would, however, make removal and subsequent refurbishment of components an easier task by slowing down corrosion.

An evaluation of refurbishment capability is provided in summary form in Table 32.

OFF-NOMINAL CASE RENOVATION EVALUATION

The following paragraphs discuss some of the off-nominal cases where system refurbishment may be accomplished:

The recovery system features full redundancy in drogue and main parachute subsystems. A single operating drogue parachute will meet all the applicable requirements of stabilization and deceleration of the command module. A probable cause of a single drogue deployment failure would be failure of the pyroinitiation in one of the mortars. Refurbishment of the unfired mortar is possible, providing the cause of malfunction can be definitely pinpointed. Procedure for refurbishment has been described in previous discussions.

Two main parachutes will provide the command module with a landing velocity which is within the capability of an earth-landing system. The probable cause of a chute deployment failure is failure of the pilot parachute mortar. In this case, the main pack will be retained on the parachute deck. The parachute pack cannot be reused, however, because of deterioration from sea water and the unknown problems associated with reuse after a second time in space.

Table 32. Docking Recovery and Earth-Landing System Summary of Refurbishment Evaluation

Component	Identification	Status at Completion of Mission	Degree of Degradation Due to Space Environment	Degree of Degradation Due to Operational Use	Refurbishment/Replacement Assessment
DOCKING SYSTEM					
Probe Mech Docking Ring	V36-575-101 V36-316050 V36-551300	Jettisoned prior to entry	NA	NA	Replace total system
RECOVERY SYSTEM					
Drogue Parachute Mortar	ME 901-0724-0001	Parachute disconnected at altitude	Minimal degradation of mortar case	Possible damage of mortar during deployment	Replace parachute, Sabot, Lid, Refurbish mortar, as required, if undamaged
Main Parachute	ME 901-0695-001	Parachute lost or submerged	NA	Major fabric degradation	Replace
Retention Assembly	ME 901-0693-0001 0693-0002		NA	Damaged	Replace
Pilot Mortar	ME 623-0001-0001		Minimal degradation of mortar case	Possible damage of mortar during deployment	Replace parachute, Sabot, Lid, Refurbish mortar, as required
Main Parachute Riser	ME 901-0694-001	Could be cut after landing	NA	Sea-water intrusion into cable strands	Replace
Drogue Parachute Riser	ME 453-00051-0091	Severed during operation	NA	NA	Replace
Sequence Controller	ME 901-0001-0019		Minimal	Relay life approaching limit	Replace
Pyro Controller Verification Box	V16-540130		Minimal	Relay life approaching limit	Replace
Parachute Attachment Fitting	V36-596-002		Minimal	Possible scuffing of surface	Remove scuff marks: Refurbish, as required
EARTH-LANDING SYSTEM					
Crew Couches	V16-531-501		Minimal	Structure - minimal. Upholstery, harness: possible abrasion and general deterioration	Refurbish as required
Couch Attenuation Struts	V16-571-411 V16-571-412 V16-571-314 V16-571-016	Stroking unlikely in nominal landing case	Minimal	Minimal in nominal landing case	Refurbish as required
Compressor (Uprighting System)	ME 281-0020	NA	Minimal	Slight wear and deterioration	Refurbish, as required
Bag and Stowage (Uprighting System)	ME 192-0042	Inflated	Minimal	Slight wear and abrasion	Replace bags, if activated
Solenoid Valve (Uprighting System)	ME 284-0285	NA	Minimal	Low-level of deterioration	Refurbish as required
Pressure Relief Valve (Uprighting System)	ME 284-0314	NA	Minimal	Low-level of deterioration	Refurbish as required
Check Valve (Uprighting System)	ME 284-0283	NA	Minimal	Low-level of deterioration	Refurbish as required

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XI. GUIDANCE AND NAVIGATION

The guidance and navigation subsystem forms the primary guidance and navigation control system (PGNCS) for the spacecraft (Figure 59). The three primary functions of the PGNCS are:

1. Maintenance of an internal reference, which is used as a basis for measurements and computations
2. Calculation of the position and velocity of the spacecraft
3. Generation of attitude error signals and thrust commands necessary to maintain the required spacecraft trajectory.

CONDITION OF RECOVERED SUBSYSTEM

Degradation of the PGNCS due to orbital operation is limited to the specific assemblies described in the following paragraphs.

Inertial Measurement Unit (IMU)

The IMU (Figure 60) provides an inertial reference consisting of a stable member gimbaled for three degrees of freedom and stabilized by three integrating gyros and associated servos. Acceleration of the spacecraft is sensed by the pulsed integrating pendulum accelerometers (PIPA) mounted orthogonally on the stable member.

The degradation of the IMU, during orbit, is limited to increasing drifts due to bearing wear in the gyro assemblies and gimbal mounts. Run-time on the IMU is expected to be about 1400 hours at the end of a mission. The gyro end-life is a function of bearing manufacture, gyro assembly, and operating conditions. The gyro end-life is not a readily predictable quantity. In discussions, on the life expectancy of the IMU gyros John E. Miller, Deputy Associate Director of the Instrumentation Laboratory at Massachusetts Institute of Technology, stated that failures have occurred in as little as 200 hours of gyro operation while other gyros were operating following 5000 hours of use. Mr. Miller recommended that gyro testing to the Apollo operational checkout procedures to be used as gyro replacement criteria.

Loss of power to the IMU heaters during landing and post recovery with subsequent cooling below plus 50 degrees F could cause damping fluid changes in the PIPA's (Reference 6). This change in damping fluid is not destructive, but will require that the PIPA's be recalibrated.

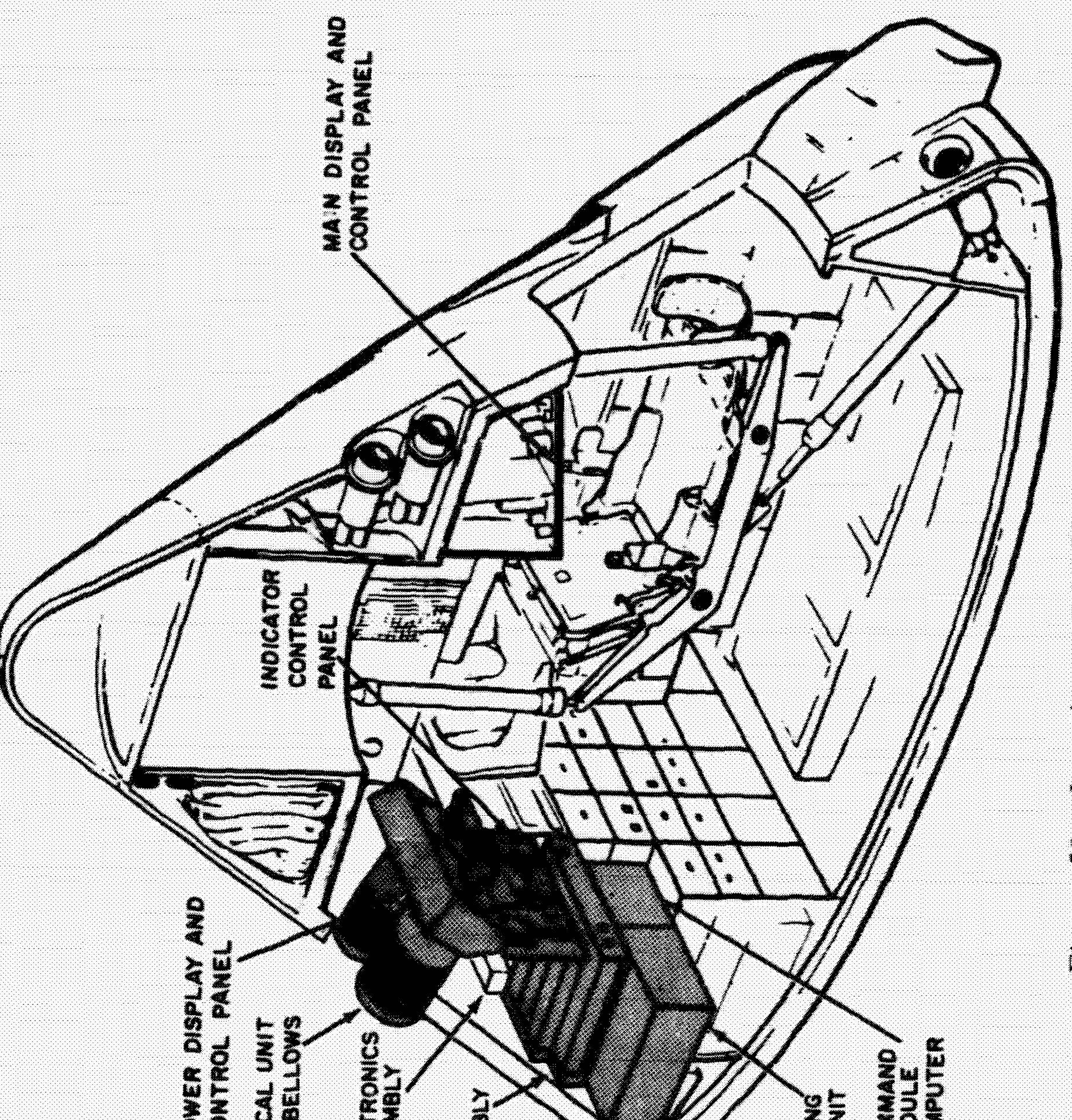


Figure 59. Location of PGNCS Equipment

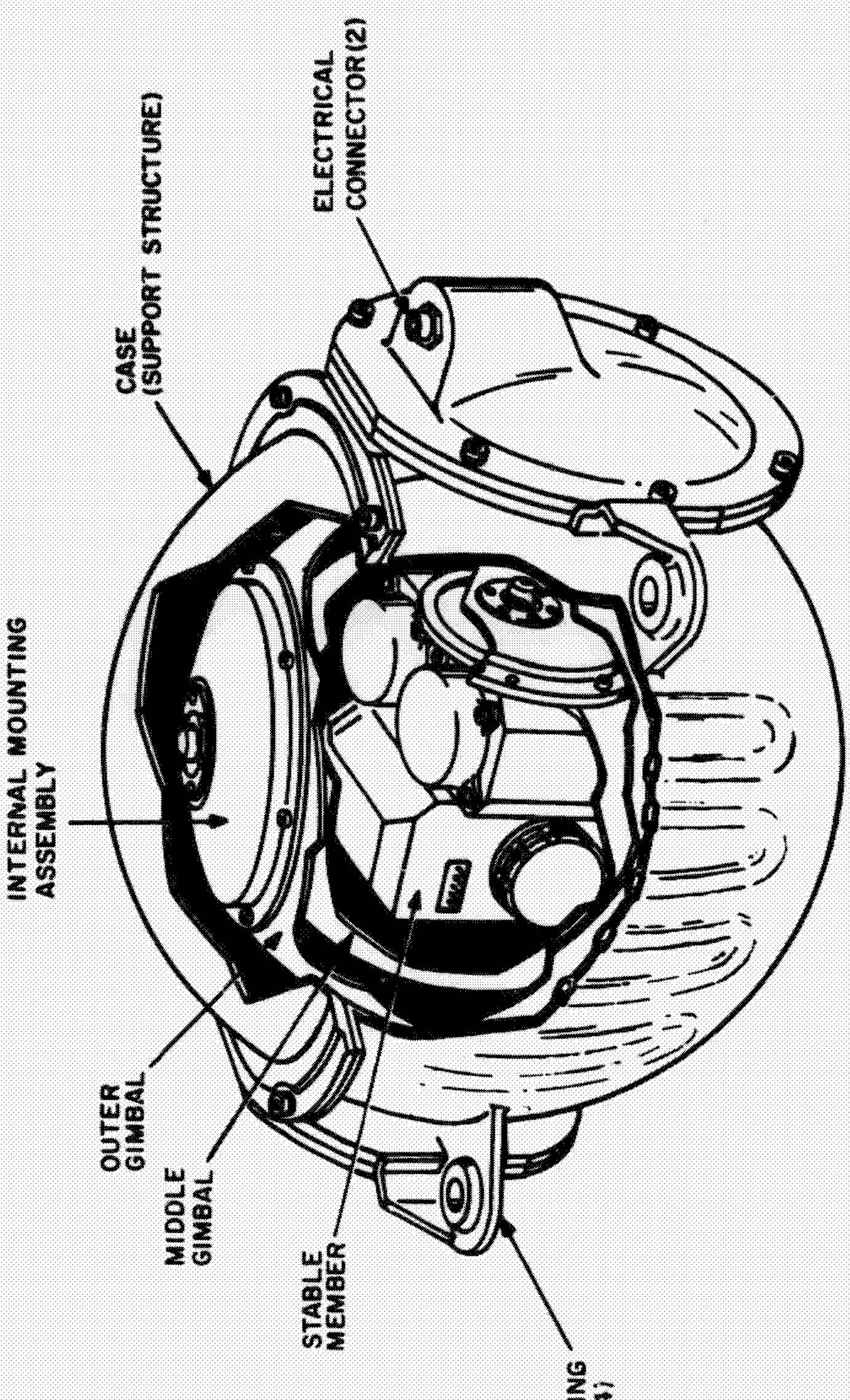


Figure 60. Inertial Measurement Unit

Measurement of the isolator lockout receptables on Spacecraft 011 (Reference 7) gave no indication of appreciable change from the initial installation in orientation of the navigation base upon which the IMU and optical assembly are mounted.

Optical Assembly

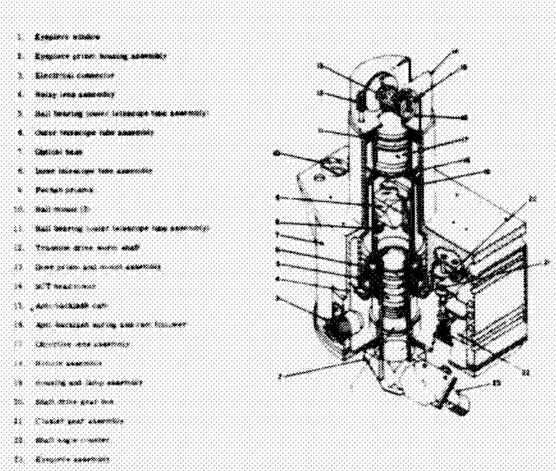
The optical assembly is used to determine the position of the spacecraft and the orientation of the IMU in space and contains a sextant and scanning telescope (Figure 61). The sextant is a high-magnification dual line-of-sight device used for precision angular measurements. The telescope has a wide field of view, a single line of sight, and can be used for coarse acquisition of stars and landmarks and orbital tracking of landmarks.

The optical assembly is presently exposed to the exterior environment surrounding the Command Module. On May 4, 1966, MCR 1491 was generated, at the request of NASA, removing the NAA optics door and replacing it with two cone-shaped heat shields (Reference 8). The heat shields, while preventing heating of the command module interior via the optical assembly, have no provisions for protecting the optical mirrors and prisms from contamination. The condition of the Spacecraft 011 optic windows is shown in Figures 62 and 63. The damage shown can come from three causes: (1) during orbit, the solar wind flux (Reference 9) can degrade the non-reflective coatings and can cause lens damage; (2) he reentry heating of the main heat shield can cause sputtering deposits on the optic mirrors and prisms (Figures 62 and 63); and, (3) splashdown contamination by salt water will over-shadow the above degradation as there is no provision for the prevention of salt water entry into a major portion of the optical assembly.

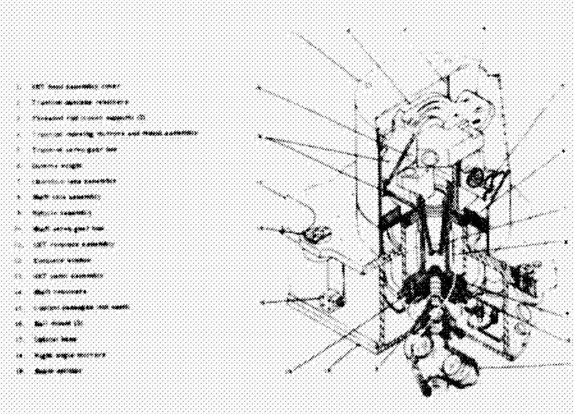
As shown in Figure 63, both the scanning telescope cover and the sextant cover have openings to allow the prisms and mirrors to look directly into space. Salt water will enter this area and coat and corrode all exposed parts. Corrosion will also occur between the bellows surrounding the sextant and telescope and the spacecraft. See Figures 64 and 65. Figure 64 shows the sextant and telescope removed from the spacecraft and still attached to the navigation base. Figure 65 shows the inside of the spacecraft at the point where the sextant and telescope bellows attach to the spacecraft wall.

Display and Keyboard Assembly (DSKY)

The DSKY assemblies form the link between the crewman and the PGNCS. Selection of a computer program or insertion of data into the guidance computer can be performed by the astronaut via the DSKY keyboard. Lamps on the DSKY indicate the program being used by the guidance computer or the results of calculation. The DSKY contains electroluminescent (EL) lamps of two types. White phosphorous EL lamps are used to illuminate the DSKY



(A) Scanning Telescope



(B) Sextant

Figure 61. Guidance and Navigation Optical Assemblies

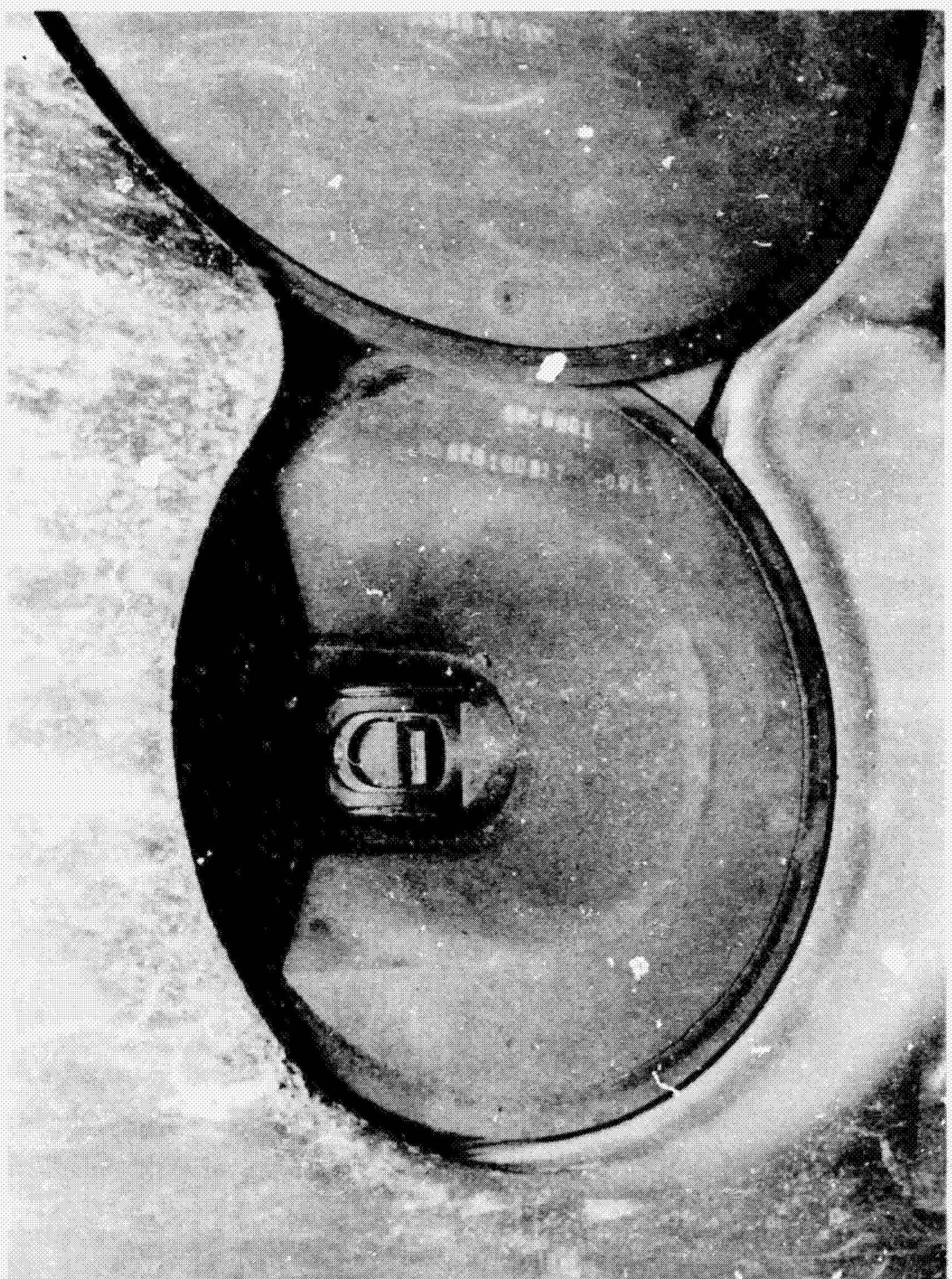


Figure 62. Spacecraft 011 Scanning Telescope, Exposed Surfaces

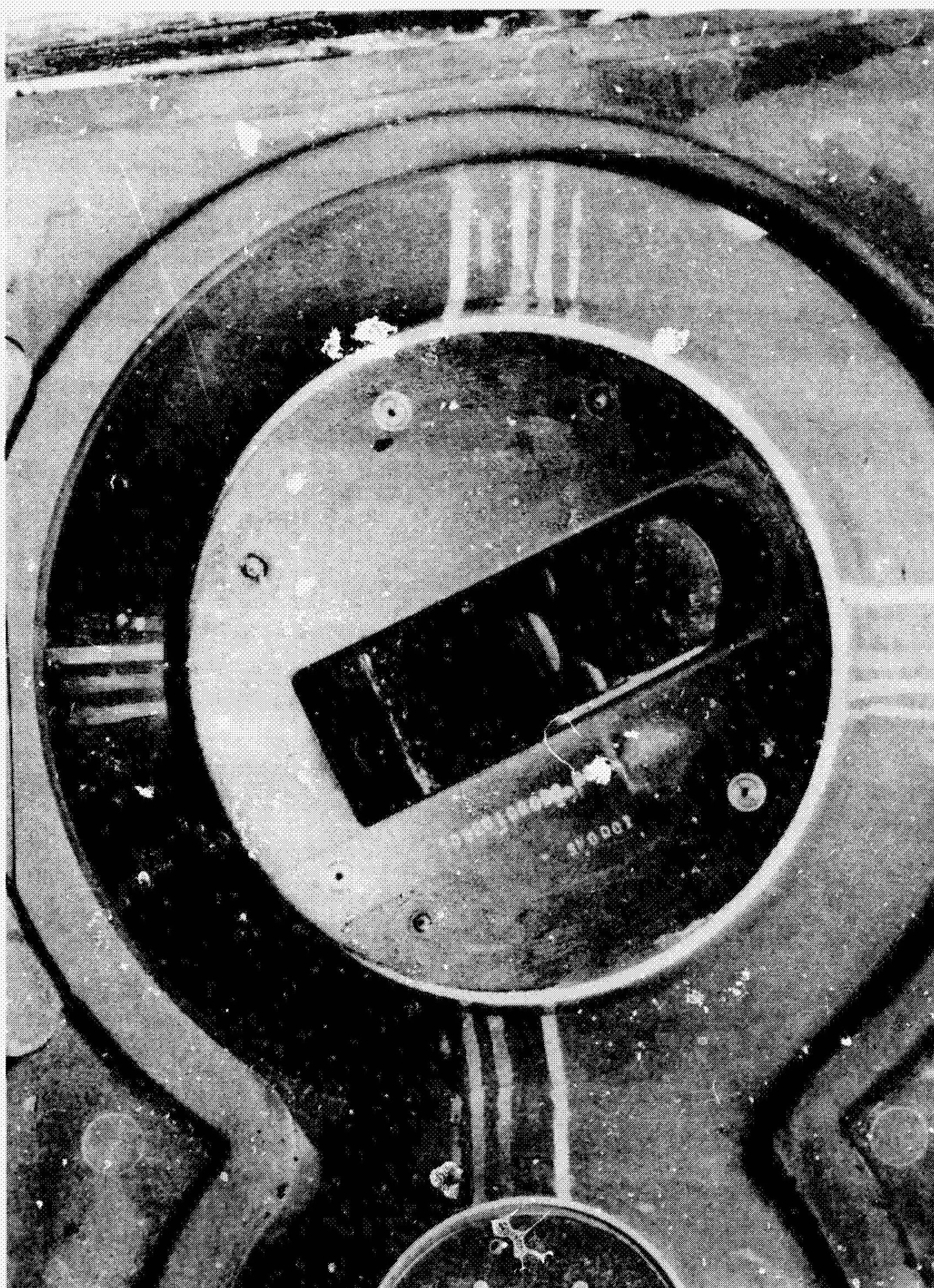


Figure 63. Spacecraft 011 Sextant, Exposed Surfaces

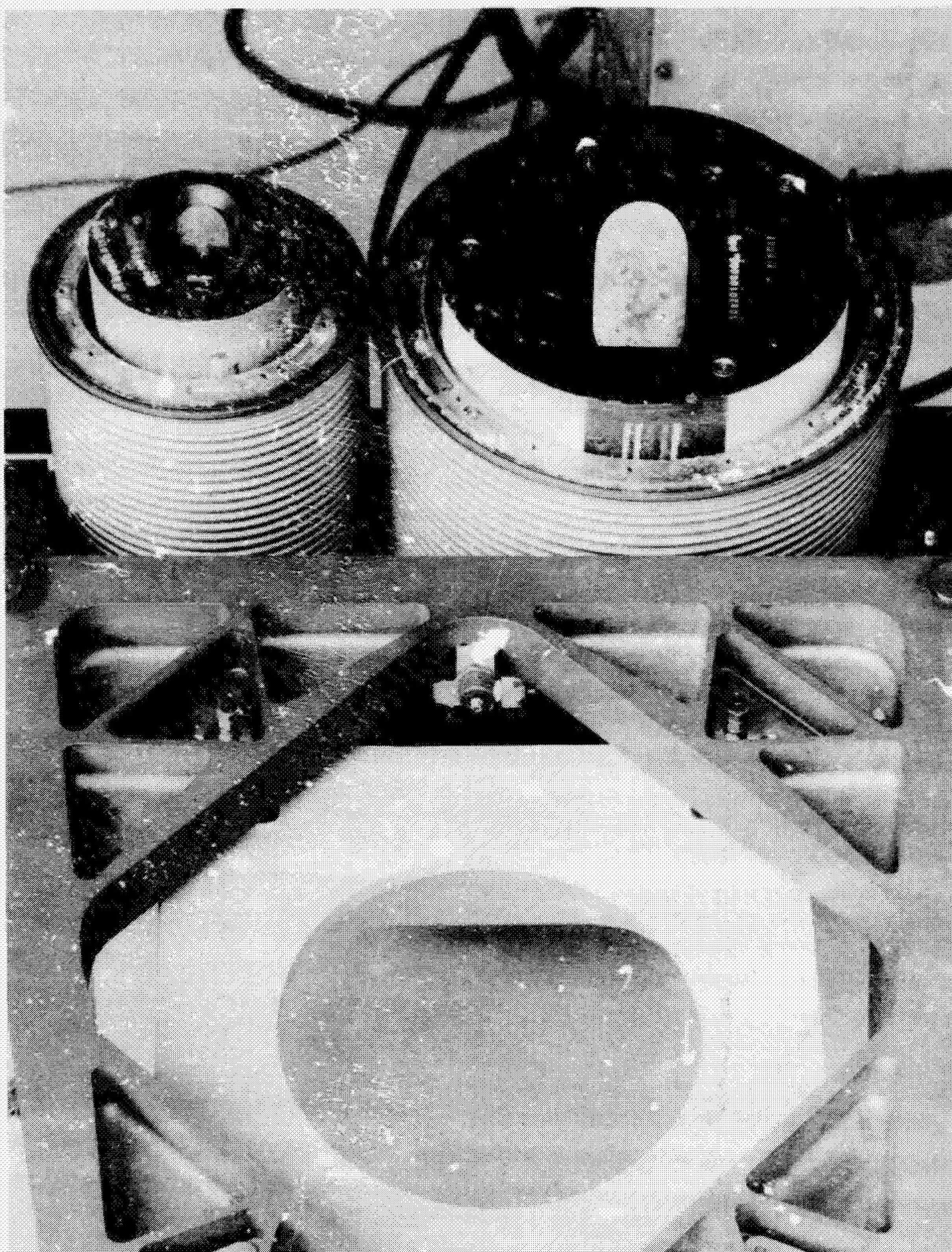


Figure 64. Spacecraft 011 Optical Assembly

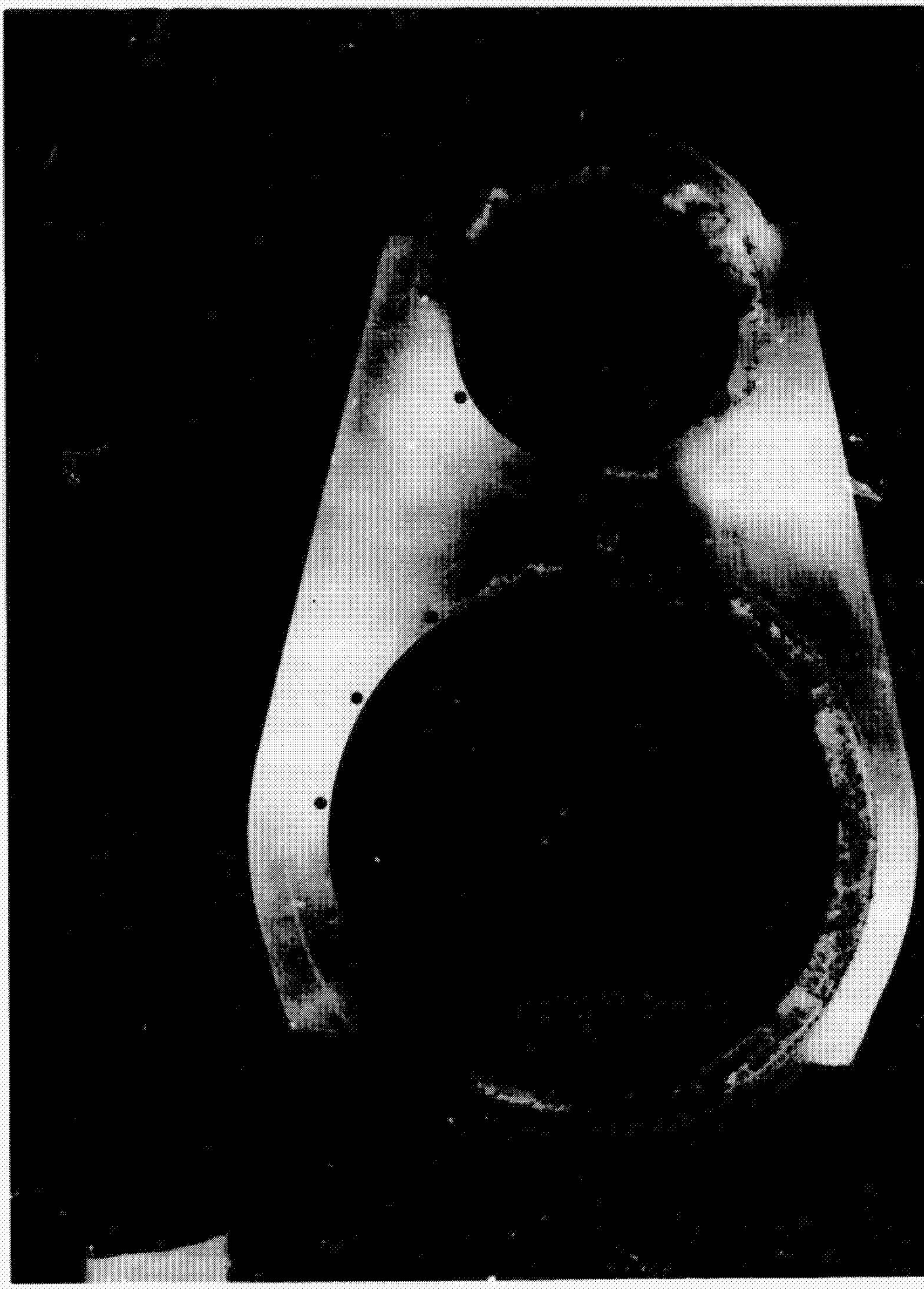


Figure 65. Spacecraft 011 Guidance and Navigation Mounting Surfaces

pushbuttons. These lamps have a high loss of illumination rate and, therefore, a short life. Illumination of the white EL lamps is down 82 percent after 2000 hours use (Reference 10).

Blue phosphor EL lamps are used to display numeric information on the DSKY (+, 0, 1, 2, 3, etc.). The blue EL lamps have a slower brightness decay rate that the white EL lamps. Illumination of the blue EL lamps is down 65 percent after 2000 hours used. In Spacecraft 011, the verb electroluminescent display on the navigation DSKY had a 3/4-inch crack in the upper right corner. (See Section VIII for further fiscussion of EL lamp characteristics.)

Command Module Computer (CMC)

The CMC performs space flight data handling and computations. The CMC is composed of a general-purpose digital computer employing a core memory (Figure 66). The fixed-core memory portion of the CMC is not degraded during flight, but must be replaced as it contains the mission flight dynamics, which are applicable only to specific missions, and the celestial navigation coordinates that are a function of time.

Associated Equipment

The condition of the following equipment is expected to be within operational tolerance following a mission: power and servo assembly, coupling data unit, and indicator control panel.

RENOVATION REQUIREMENTS

Inertial Measurement Unit (IMU)

The IMU shall be checked for gyro drift and shall have any gyro exceeding the requirements of the Apollo operational checkout procedure replaced. The IMU shall be functionally checked and then stored. Upon the required activation time, before flight, the IMU shall be brought up to temperature and the PIPA calibration performed.

Optical Assembly

The optical assembly shall be dismantled and all components checked for contamination and transmissibility. The optical assembly shall then be assembled and adjusted, per specification, with all seals and degraded parts being replaced.

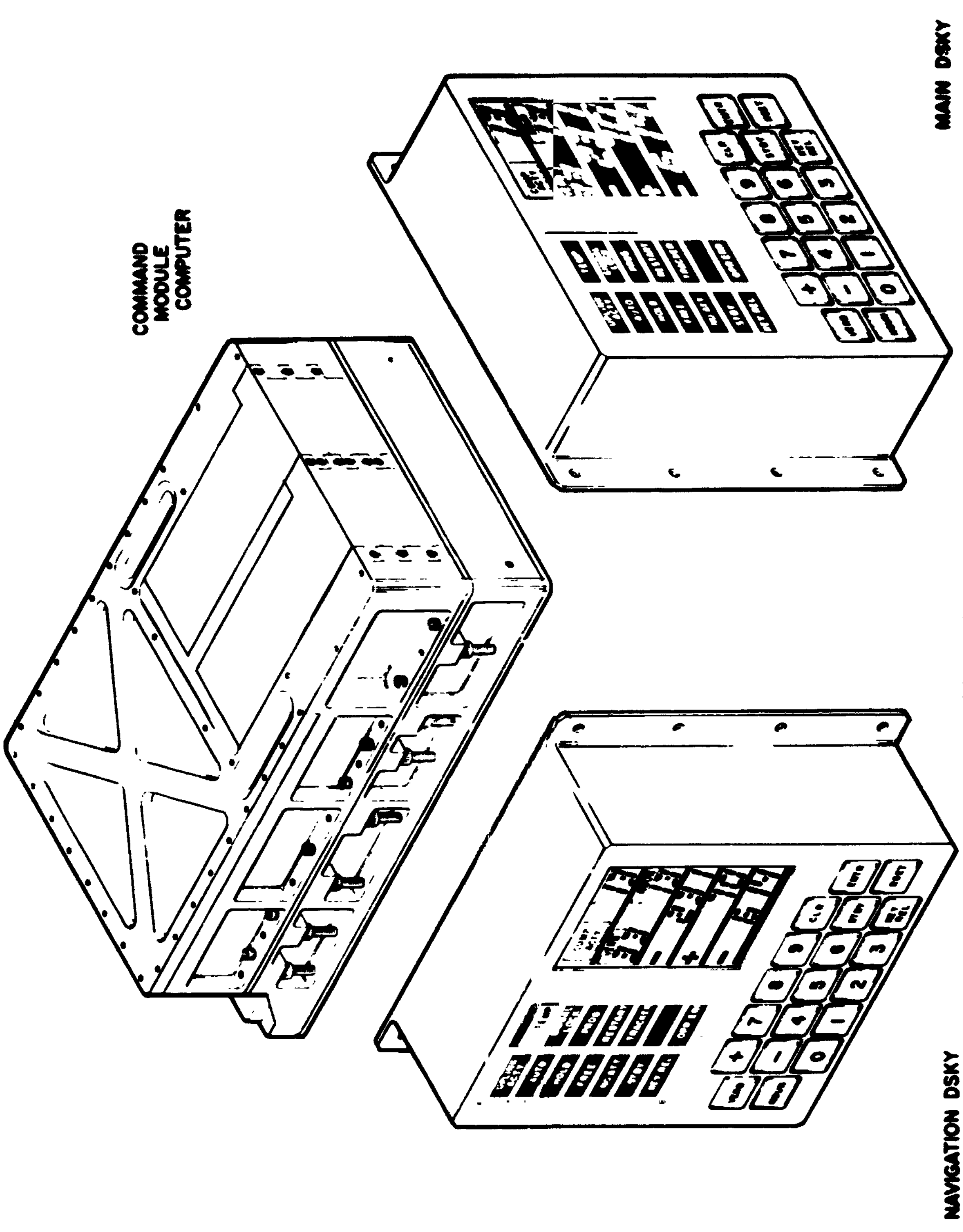


Figure 66. CSS Equipment

Display and Keyboard (DSKY)

The DSKY shall be dismantled and all subassemblies checked to specification. The electroluminescent displays and keys shall be tested then reused or replaced. The DSKY shall then be assembled, per specification, with all seals and degraded parts being replaced.

Command Module Computer (CMC)

The CMC shall be tested to specification and the appropriate mission memory shall be installed.

Associated Equipment

The associated equipment and harnesses shall be checked to specification prior to installation with the renovated assemblies.

Guidance and Navigation System

The guidance and navigation system shall be system-tested prior to installation in the command module.

POSTRECOVERY OPERATION REQUIREMENTS AND CONSTRAINTS

Upon recovery of the spacecraft, AC Electronics (Reference 11) has recommended that heater powers be reconnected to the IMU assembly. The addition of heater power should be accomplished by the use of GSE equipment. AC Electronics' recommendation states that the IMU should not be allowed to go below +65 degrees F. An investigation of flown IMU hardware, upon which the temperature level has not been maintained, should be conducted to ascertain if temperature control is mandatory or if recalibration will overcome the cooling effects on the inertial sensors.

Upon recovery of the spacecraft, a complete washing of the exterior parts of the optical assembly shall be performed using demineralized water. At the earliest convenience, the optical assembly shall be removed and the area between the command module wall and the optical assembly bellows shall be washed with demineralized water and treated as required to prevent continued salt-water corrosion.

EXPLORATORY TEST REQUIREMENTS

Checkout of the IMU on the BME for catastrophic failure of any of the inertial sensing components is recommended. All other failures, except for the known optical degradation, should be minor in nature and will be covered in normal renovation.

XII. STABILIZATION AND CONTROL

The stabilization and control system (SCS) provides stabilization and control of the vehicle during rotational, translational, thrust maneuvers and attitude hold. The SCS also functions as a strap-down guidance system. The system is used as a backup for the primary guidance and navigation subsystem. The assemblies are depicted in Figure 67.

CONDITION OF RECOVERED SUBSYSTEM

A discussion of the anticipated degradation of the SCS performance capability due to preflight, flight and postrecovery operations is provided below by end item.

Gyro Assembly (Two Each Per System)

The gyro assemblies include three single degree-of-freedom rate-integrating gyros and the necessary electronics to provide output signals proportional to either angular displacement or angular rate. It is expected that the three gyro units in each end item (six gyros total) will be degraded because of system operation in preflight test and flight operation. The degradation is due primarily to worn bearings. The gyro assemblies have a maximum life of 2500 hours. Figure 68 depicts a typical integrating gyro.

Gyro Display Coupler (GDC)

The GDC provides interface between the gyro assembly and the command module computer and the flight director attitude indicator (DSAI).

The GDC is expected to experience performance degradation in the electromechanical stepper motor-resolver assemblies as a result of wear in gear trains.

Display Electronic Assembly (DEA)

The DEA provides the interface between the SCS displays and other components of the guidance and control system.

The DEA is an electronic device and should be degraded only because of operating time incurred during preflight testing and flight operations.

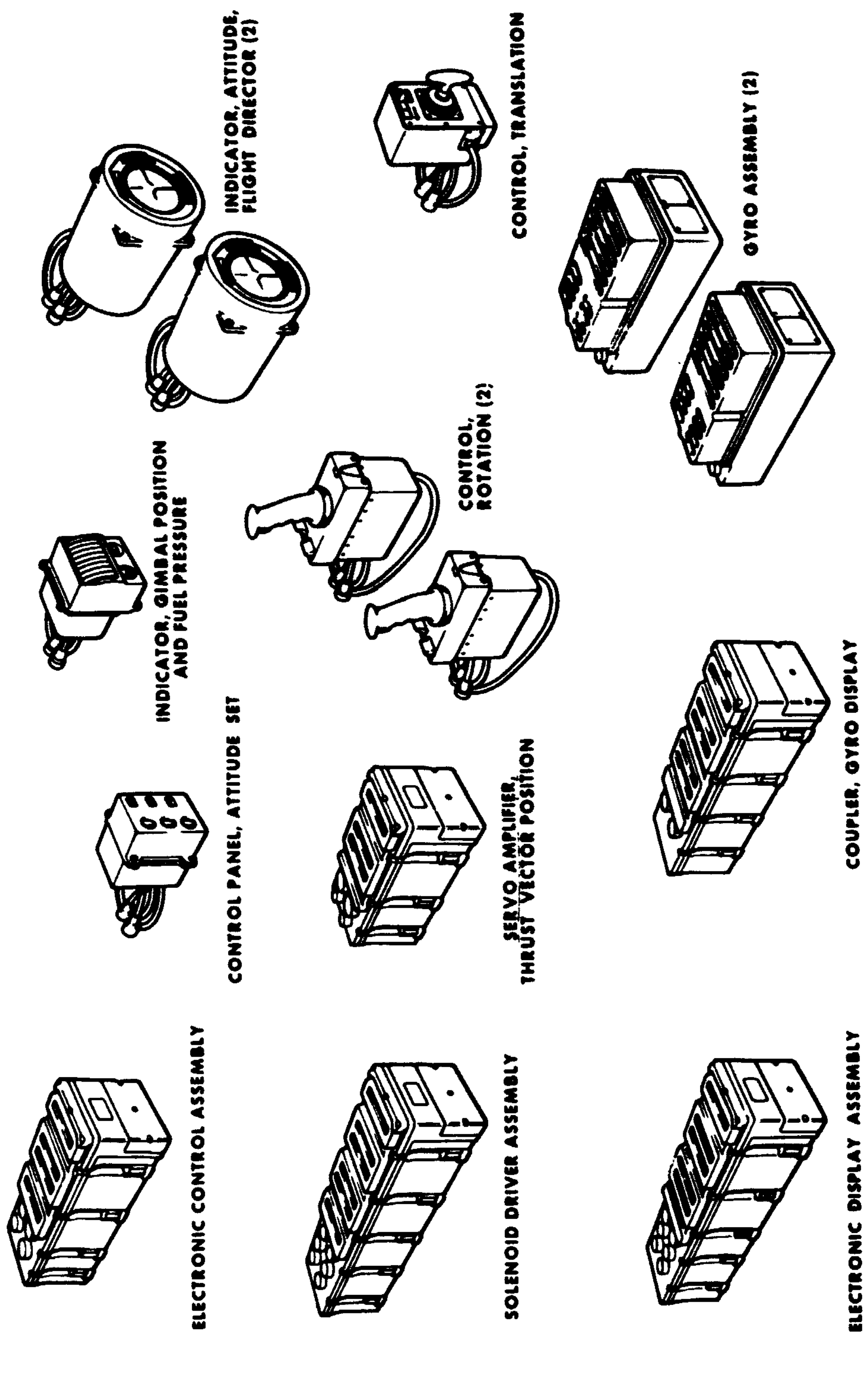


Figure 67. SCS Flight Hardware, Block II

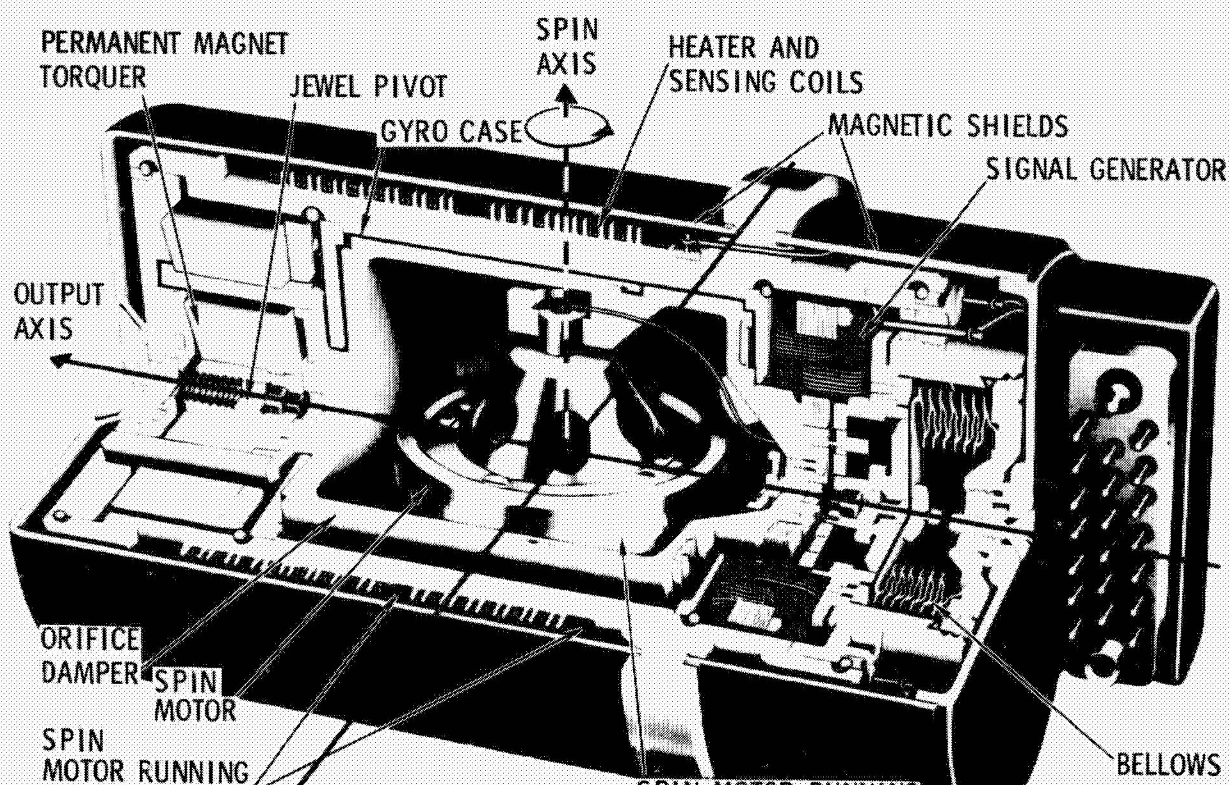


Figure 68. Typical Floated Integrating Gyro

Flight Director Attitude Indicator (FDAI) (Two each per system)

The FDAI displays attitude, attitude errors and rate information.

The performance of the FDAI is expected to be degraded in the area of the electromechanical drive assemblies because of wear in the gearing and slip-rings.

Control Electronic Assembly (CEA)

The CEA performs summing, shaping, and switching functions of the sensor and manual input signals necessary to maintain stabilization and control in all three axes.

The CEA is an electronic assembly and should be degraded only because of operating time incurred during preflight testing and flight operations.

Thrust Vector Position Servo Amplifier (SAA)

The SAA controls the service propulsion system actuator assembly.

The SAA is an electronic assembly and should be degraded only because of operating time incurred during preflight testing and flight operation.

Solenoid Driver Assembly (SDA)

The SDA controls the on-off function of the RCS and SPS valves.

Attitude Set Control

The attitude set control provides the capability to manually set desired attitudes for use in the attitude reference system.

The device is an electromechanical assembly and may incur slight mechanical wear during preflight testing and flight operation.

Gimbal Position/Fuel Pressure Indicator

The indicator provides a display of the angular positions of the pitch and yaw SPS engine gimbals or of the S-II/S-IVB fuel pressure. The panel also includes controls for manually positioning the SPS engine gimbals.

The device could incur performance degradation because of mechanical wear of the gimbal position set control incurred during preflight testing and flight operations.

Rotation Control (Two each per system)

The rotation control is a hand-operated device which provides rotational control of the vehicle.

The device is an electromechanical assembly and can be expected to incur performance degradation because of mechanical wear of the transducers and control linkage during preflight test and flight operations.

Translation Control

The translation control is a hand-operated device which provides control of vehicle translation in three planes.

The device consists of a set of switches actuated by a single control handle. The device can be expected to incur mechanical wear in the control linkage during preflight tests and flight operations.

RENOVATION REQUIREMENTS

Each electronic and electromechanical assembly shall be tested at the end-item level. Any out of tolerance indication or malfunction shall be investigated and repaired.

It may be possible to repair electronic assemblies which indicate an out of tolerance condition by changing the gain and trim components. These components are located on top of the interconnect matrix assembly (Figure 69). A malfunction within a module will require removal of the cordwood-stacked module assembly.

Electromechanical assemblies such as the FDAI, Rotation Control etc., may require replacement of gear trains and transducers because of wear.

It is recommended that the gyro units of the gyro assembly be replaced. This recommendation is based on an estimated 1400 hours of accumulated operating time accrued during preflight test and flight operation for the first mission.

Each assembly should be purged and recharged in accordance with current specifications.

EXPLORATORY TEST REQUIREMENTS

The two gyro assemblies should be tested on the BME for catastrophic failures which may have occurred in the gyro units.

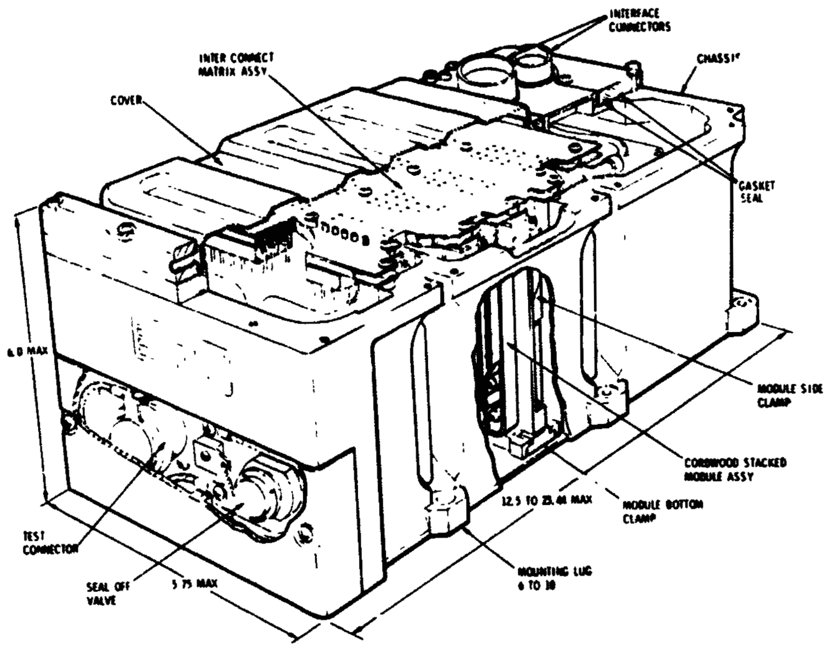


Figure 69. Typical Block II Electronic Assembly

XIII. ENTRY MONITOR SUBSYSTEM

The entry monitor subsystem provides a backup to the guidance and navigation (G&N) or SCS subsystem during thrusting and entry maneuvers. The entry monitor subsystem performs the following functions (Figure 70): (1) g-velocity entry trace for g onset evaluation; (2) corridor evaluation; (3) 0.05 g detection; (4) lift vector direction; (5) range to go; (6) delta V measurements.

CONDITION OF RECOVERED SUBSYSTEM

The operational environment during a normal mission should have little effect on the subsystem. The subsystem is sealed for protection from the environment and is located in the main display panel which is not subject to adverse environmental conditions. The entry monitor subsystem electronic assembly is a low-use item, used only during thrusting or entry, and is expected to be operated for only thirty hours during a mission. The scroll and servo assembly, used during entry, is expected to operate for only two hours during a mission (Figure 70).

RENOVATION REQUIREMENTS

The scroll in the scroll assembly must be replaced as the corridor limits vary from mission to mission and a permanent trace is scribed upon the surface of the scroll during each mission (Figure 71). The electroluminescent displays should be checked for light output to determine if replacement is required.

FUNCTION

BACKUP TO G&N AND SCS

- G-V ENTRY TRACE
- CORRIDOR EVALUATION
- 0.05 G DETECTION
- LIFT VECTOR DIRECTION
- RANGE TO GO
- ΔV MEASUREMENT

OPERATIONAL DEGRADATION

- NONE ANTICIPATED
- SCROLL REPLACEMENT

ESTIMATED RENOVATION COST

~ 5% OF NEW SUSTEM

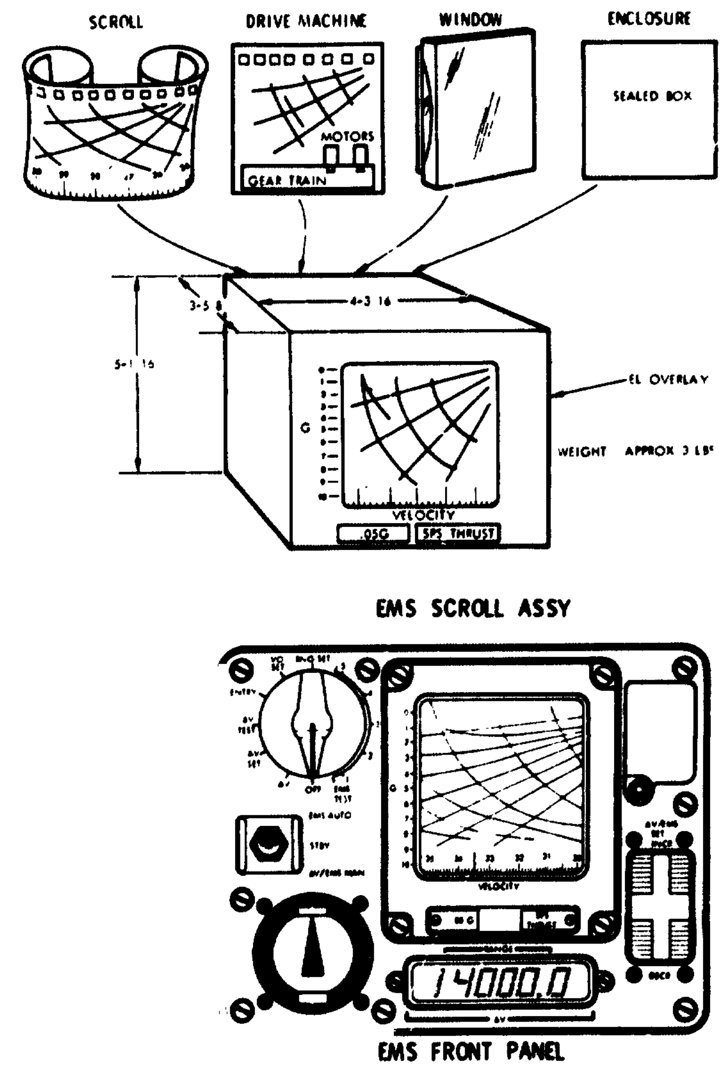


Figure 70. Entry Monitor System

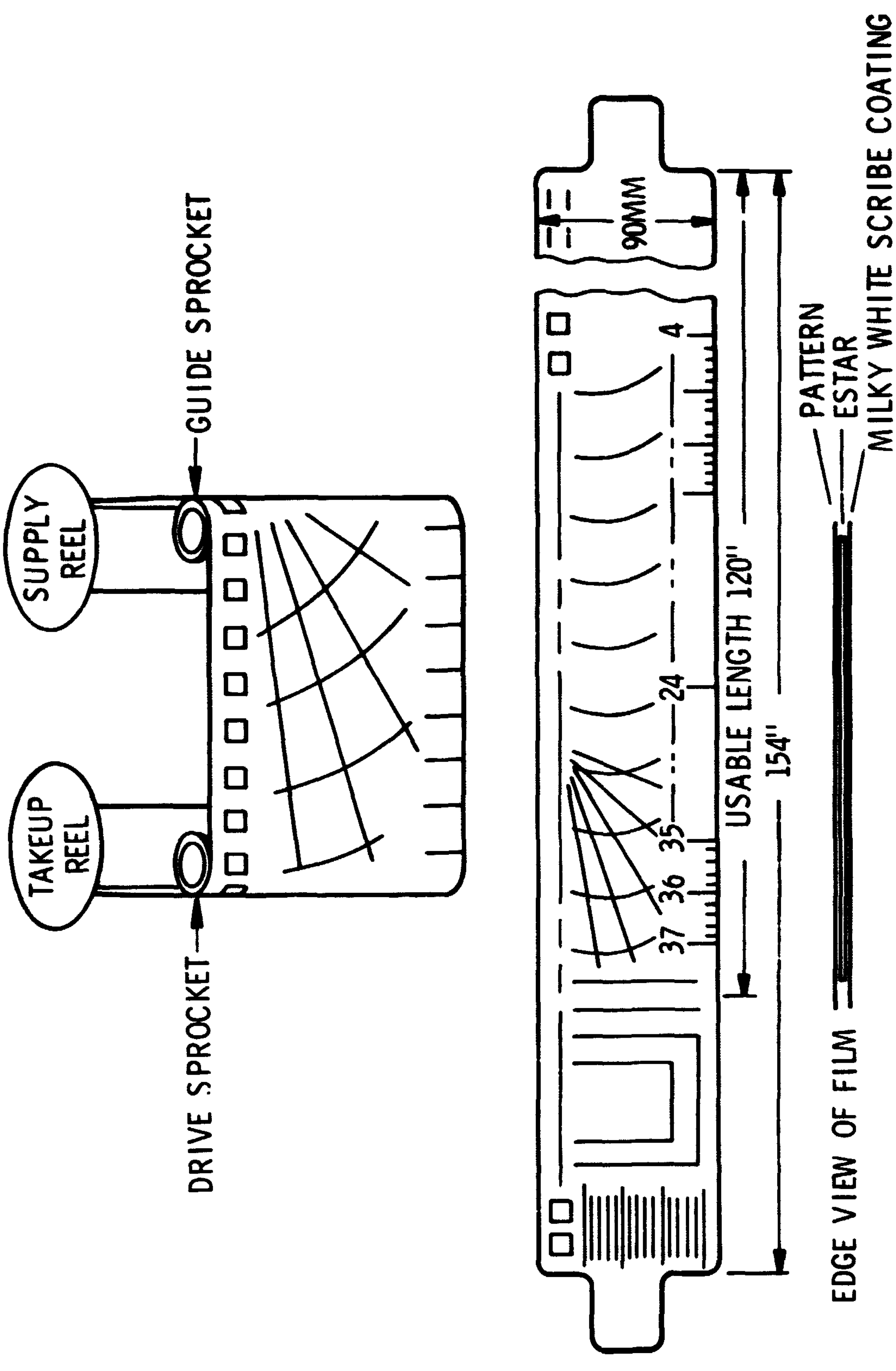


Figure 71. EMS Scroll and Scribe

XIV. REACTION CONTROL SYSTEM

CONDITION OF RECOVERED SUBSYSTEM

During the course of an Apollo mission, potential degradation of the command module reaction control system (CM RCS) performance capability can result from a number of conditions. These conditions are identified as follows:

1. Exposure to boost heating
2. Exposure to structural and dynamic loading during boost
3. Exposure to space vacuum
4. Exposure to radiation
5. Exposure to space temperature extremes
6. Exposure to meteoroid impingement
7. System operation and propellant exposure
8. Exposure to reentry heating
9. Structural and dynamic loading during landing
10. Exposure to sea water
11. Postlanding recovery operations

For a normal Apollo 14-day lunar mission, no evidence exists to indicate that any CM RCS component (Table 33) will be degraded by conditions of 1, 2, 3, 4, and 6. For longer mission durations (45 days and up), however, certain unknowns exist with respect to the capability of some RCS components to withstand the conditions of 3 and 4. Longer mission durations necessitate exploratory tests of the following to determine degree of degradation:

Table 33. CM RCS Components

Component	Identification	Supplier
Oxidizer tank	ME282-0006	Bell Aerosystems Corp.
Fuel tank	ME282-0007	Bell Aerosystems Corp.
Helium tank	ME282-0002	Menasco Manufacturing Co.
Helium fill coupling	ME273-0010	On Mark Couplings, Inc.
Test point coupling	ME144-0023	Lear Siegler, Inc.
Propellant couplings	ME273-0011 -0019 -0021 -0024	J.C. Carter Co.
Helium explosive valve	ME284-0019	Pelmec
Propellant explosive valve	ME284-0130	Pelmec
Helium pressure regulator	ME284-0022	Fairchild Strates Corp.
Check valve assembly	ME284-0024	Accessory Products Co.
Helium pressure relief valve	ME284-0062	Calmec Mfg. Corp.
Burst diaphragm isolation valve	ME251-0005	Pyrodyne, Inc.
Propellant latching solenoid valve	ME284-0276	National Water Lift Co.
Flexible hose	ME271-0019	Titeflex
Dynatube fitting	ME273-0046	Resistoflex
Rocket engine and nozzle extension	ME901-0067 ME901-0189	Rocketdyne

Component	Potential Problem
Pressure relief valve	Vacuum cold welding
Pressure regulator	Vacuum exposure on poppet and seal materials
Oxidizer tank	Radiation exposure of bladder under oxidizing conditions
Engine valve seat	Radiation

The following paragraphs discuss the other conditions which potentially result in a degrading effect on CM RCS components.

Space Temperature Extremes

Under certain mission conditions, it is possible for RCS components in the CM aft equipment toroidal section to see temperature extremes of 15 F to +200 F. No problem is expected for the components at the low temperature; if temperatures reach 200 F or above, seals and potting compound in the following may be degraded:

Helium fill couplings
Test point couplings
Helium pressure regulators
Check valves
Propellant disconnect couplings
Burst diaphragm isolation valves
Propellant isolation valves

System Operation and Propellant Exposure

In normal system usage, some components of the CM RCS operate in such a manner that they cannot be used again without replacement or refurbishment. In addition, some components are limited life items whose number of life cycles may be exceeded during a combination of checkout and mission usage. The CM RCS components in this category are as given in Table 34. The rocket engines/nozzle extensions are also limited life items because of their ablative design. The actual replacement point for the items in Table 34 will vary with the type and duration of the mission; where applicable this replacement point includes a safety factor of 10 percent of the limited life of the hardware. From the operational standpoint, and from Table 34, the following CM RCS components will be degraded from a normal Apollo mission:

Table 34. CM RCS Limited Life Components

Name	Part No.	Limited Life (Cycles)	Replacement Point (Cycles)		Remarks
			Earth/LOR Orbit	Ground Test Suborbit Supercircular	
CM oxidizer tank	ME282-0006	3000**	2364	2700	Pressure cycles
Tank bladder		Wet 20***	19***	19***	Bladder cycles
		Dry 40	36	36	
CM fuel tank	ME282-0007	3000**	2364	2700	Pressure cycles
Tank bladder		Wet 20	19	19	Bladder cycles
		Dry 40	36	36	
CM helium tank	ME282-0002	3000	2700	2700	Pressure cycles
Burst disc assembly	ME251-0005	800*	790	790	
Filter and burst diaphragm			40	40	
CM engine oxidizer injector valve	(ME901-0067)	200 Dry*	200 Dry	200 Dry	
CM engine fuel injector valve	(ME901-0067)	200 Dry*	200 Dry	200 Dry	

*Life of assembly body only
**Cycles of titanium shell without bladder
***Only qualified to 6 (wet) expulsion cycles

		Recommendation
1.	RCS engines/nozzle extensions	Replacement necessary
2.	Oxidizer and fuel propellant tank bladders	Replacement necessary
3.	Helium explosive valves	Replace
4.	Propellant explosive valves	Replace o1 :s actuated in mission
5.	Burst diaphragm isolation valves	Refurbish
6.	Helium relief valves	Replace if diaphragm ruptured
7.	Oxidizer fill and vent couplings	Replace because of potential chemical attack
8.	Oxidizer latching solenoid valves	Require careful retest or replacement

Reentry Heating

Maximum temperatures expected for CM RCS components due to reentry plus soakback heating will potentially degrade seals and/or potting compounds in some components. If temperatures do not exceed approximately 200 F, no problems are expected. However, for excessive temperatures, the following components may be degraded:

Helium fill couplings
Test point couplings
Helium pressure regulators
Check valves
Propellant disconnect couplings
Burst diaphragm isolation valves
Propellant isolation valves

Also, the ablative rocket engine/nozzle extensions will be significantly affected by reentry heating in some locations on the CM.

Structural and Dynamic Loading During Landing

The CM RCS of S/C 009 and 011 appeared to be in excellent condition following the impact of water landing. On a normal mission and landing, this is expected to be the case. However, there is a possibility of damage to RCS components in the case of the most adverse landing shocks, especially to those components located near the impact area. The following components are especially susceptible to this condition:

Oxidizer tanks
Fuel tanks
Helium tanks
Propellant solenoid valves
Rocket engines/nozzle extensions

Water Damage

Salt water corrosion has a potentially degrading effect on all CM RCS components and plumbing (especially brazed joints). S/C 009 had evidence of substantial corrosion; S/C 011 was much better than 009 but still had salt deposits on the bottom of tanks and lines. If procedures are implemented on subsequent spacecraft requiring washdown as soon as possible after impact, corrosion may be negligible. The RCS engines/nozzle extensions, however, and possibly some component wiring may still be adversely affected by salt water and washdown water. Under this condition, all potentially reusable components and plumbing will have to be inspected, cleaned, and pressure-tested prior to assessment for reuse.

Postrecovery Operations

Current postrecovery tests and inspection are primarily designed to evaluate anomalies discovered in flight (where known), to ascertain any externally evidenced physical damage, and to discover existence of additional anomalies without functional checkout of the subsystems. In the case of the CM RCS, postrecovery decontamination of the system and possibly disassembly of components may degrade the capability of some components. In the case of S/C 011, methanol introduced into the fuel tank in postrecovery decontamination reacted with residual Freon used in preassembly tank tests, and the reactants chemically attacked the aluminum standpipe in the tank. Also, the unfortunate experience with methanol in the S/C 017 SPS tanks reveals that methanol cannot be used with titanium tanks. The Apollo program has taken action to forbid the use of Freon in the fuel side of the system and to find a substitute material (perhaps isopropanol) for the methanol. With these problems resolved, the post-recovery operations are not expected to degrade the capability of any potentially reusable CM RCS component.

Summary of Anticipated Degradation

The conditions under which potential degradation of CM RCS components can occur are summarized in Table 35. This assessment is based on mission conditions that may or may not occur; most components must, therefore, be inspected and tested after the recovery before an accurate evaluation can be made of reuse/refurbishment possibilities. S/C 011 CM RCS components looked good after recovery and testing them may give a better indication of possible reuse capabilities for some components.

CM RCS EFFECT ON HEAT SHIELD

The heat shields on S/C 009 and 011 were significantly charred in the vicinity of the CM RCS roll engine nozzle exits. If this damage occurred from normal operation of the roll jets to damp roll rates during reentry, the damage must be accepted and repaired as part of CM renovation. If, however, the damage occurred during the propellant jettison period, some analysis of the possibility of minimizing this damage is justified.

After the main parachutes are disreefed in a normal reentry, the remaining CM RCS propellant must be jettisoned to eliminate possible toxicity hazards on impact; this is done by burning the propellant through ten CM RCS engines (including the four roll engines but excluding the two upper pitch engines because they are too near the parachute lines). It would seem possible to expel fuel only from the engines, dumping the oxidizer, nitrogen tetroxide, through the oxidizer dump system as is done in the pad abort mode. Nitrogen tetroxide vapor, however, can deteriorate the nylon parachutes and lines extremely rapidly, and this potential flight safety hazard has always been considered great enough in past analyses to make the burnoff mode mandatory.

It might be possible to use only two pitch and four yaw engines for this burnoff. Heat shield damage would be reduced because the pitch and yaw jets fire at right angles to the heat shield surface. However, propellant jettison and purge must be complete before water impact, even though one-half of the CM-RCS is inoperable. This would leave only three engines (one pitch and two yaw) available for propellant burnoff, and it seems quite possible that burnoff through only three engines would consume more time than would be available during parachute descent on some missions.

Further investigation may produce some alternative that would reduce the heat shield damage near the roll engines. Pending such a result, however, it must be assumed that such damage will occur and must be repaired as part of command module renovation.

Table 35. Potential CM RCS Degradation Resulting From Conditions During a Normal Apollo Mission

Component	Conditions										
	Boost Heating	Structural & Dynamic - Launch	Vacuum	Radiation	Space Temperature Extremes	Meteoroid	System Operation & Propellant	Reentry Heating	Structural & Dynamic - Landing	Water[7]	Post Recovery Ops.
Oxidizer tank							x[3]		x[2]		
Fuel tank							x[3]		x[2]		
Helium tank									x[2]		
Helium fill coupling					x			x			
Test point coupling					x			x			
Propellant couplings					x		x[5]	x			
Helium explosive valve							x[1]	x			
Propellant explosive valve							x[1,4]				
Helium pressure regulator					x			x			
Check valve assembly					x			x			
Helium pressure relief valve							x[6]				
Burst diaphragm isolation valve					x		x[1]	x			
Propellant latching solenoid valve					x		x[5]	x	x		
Flexible hose											
Dynatube fitting											
Rocket engine and nozzle extension							x[1]	x	x		

[1]Normal operation requires replacement parts or complete replacement prior to reuse.
[2]Reusable unless severe landing impact conditions in area where located was encountered.
[3]Normally, bladders only will be degraded.
[4]In normal mission, two valves are not used and should be available for another use.
[5]Oxidizer side only.
[6]Possible diaphragm rupture.
[7]Damage to components and plumbing dependent on washdown procedures; inspection and test are necessary.

RENOVATION REQUIREMENTS

Subsystem Disassembly

The four CM RCS panel assemblies (two V36-460101 helium panel assemblies, one V36-470101 oxidizer panel assembly, one V36-480101 fuel panel assembly) will have to be removed from the spacecraft and largely disassembled to replace and/or bench-test components. The four propellant tanks must be removed for refurbishment and the two helium tanks for retest. The twelve-engine/flexible hose/dynatube fitting subassemblies must be removed for engine replacement and component retest. All that will remain are tubing runs; these will probably be tested in place, but portions will be replaced if any degradation is apparent in postflight inspection.

Component Reuse, Refurbishment and Replacement Assessment

The number of each CM RCS component that must be replaced or refurbished is assessed below based on component design, capabilities, and degradation expected.

1. ME 282-0006 oxidizer tank (Bell Aerosystems Corp., two each). The bladder expulsion tanks are potentially the least reliable components in the CM RCS. Removal and complete disassembly, cleaning, inspection, reassembly with new seals and new bladders, and a new acceptance test are considered necessary. The vendor (Bell) is best qualified to accomplish this.

2. ME 282-0007 fuel tank (Bell Aerosystems Corp., two each). Same as ME 282-0006.

3. ME 282-0002 helium tank (Menasco Mfg. Co., two each). The interior of these two titanium vessels will be protected by the seals in the fill/vent couplings and the regulator/check-valve system. A new proof-pressure test to confirm tank suitability will be required. The tanks will be removed, cleaned, and inspected.

4. ME 273-0010 helium fill disconnect coupling (On Mark Couplings, Inc., Div. of Purolator Products, two each).

   These couplings, used for fill/vent of helium, contain redundant seals, one in the sealing cap and one in the brazed-on portion. It seems unlikely that sea water will penetrate this seal. The sealing cap will be examined by NAA (and replaced if necessary). Both seals will be tested, but it seems likely that these couplings will be reuseable.

5. ME 144-0023 test point disconnect coupling (Lear Siegler, Inc., 26 each). NAA is to leak-test in place and reuse or replace. (For pricing, assume 22 will be alright and 4 will be replaced.)

6. ME 273-0011 (two each), ME 273-0019 (four each), ME 273-0021 (4 each), ME 273-0024 (two each). Propellant disconnect couplings (The J.C. Carter Co., total twelve each).

   Seals and filter cannot be inspected because coupling is a welded system. To be conservative, 100-percent replacement of airborne coupling half only is planned.

7. ME 284-0019 helium explosive valve (Pelmec Div. of Quantic Industries, ten each).

   The valve is actuated in mission and must be replaced.

8. ME 284-0130 propellant explosive valve (Pelmec Div. of Quantic Industries, four each).

   Two actuated in normal mission must be replaced. Two others are actuated only in abort and are assumed retested and found acceptable.

9. ME 284-0021 helium pressure regulator unit (Fairchild Stratos Corp., Stratos Div., Western Branch, four each).

   The CM RCS pressure regulator assemblies are under full pressure for only the short period of reentry, and they are protected from internal sea water contamination by check valves and helium fill/vent couplings. Their use for a second cycle seems feasible. The effect of exterior salt water corrosion and the possibility of contamination entering the bellows system (which is vented to space) must still be determined. NAA will remove and bench-test, since welds make disassembly difficult. (For pricing assume three are acceptable and one is replaced.)

10. ME 284-0024 check valve assembly (Accessory Products Co., Division of Textron, Inc., four each).

    The check valve duty cycle is less than that experienced in the SM RCS, and reuse certainly seems feasible. Internal sea water contamination should not reach the check valves, especially if the CM RCS is flushed soon after recovery. NAA will remove and bench-test, since welds make disassembly difficult. (For pricing assume three are acceptable and one is replaced.)

11. ME 284-0062 helium pressure relief valve (Calmec Mfg. Corp., 4 each). NAA will remove and bench-test. If diaphragm is still intact, unit should be reuseable. (For pricing purposes, assume three are acceptable and one is replaced.)

12. ME 251-0005 burst diaphragm isolation valve (Pyrodyne, Inc., 4 each). Diaphragm is burst in normal mission. It will be returned to Pyrodyne for replacement of diaphragm and seals and reacceptance.

13. ME 284-0276 propellant latching solenoid valve (National Water Lift Co., a Division of Pneumo Dynamics Corp., 4 each).

    NAA will remove and bench-test, since welds make disassembly difficult. (For pricing, assume three are acceptable and one is replaced.)

14. ME 271-0019 flexible hose (Tite Flex, 24 each).

    NAA to remove with engine, test, and reuse or replace. (For pricing, assume 18 are reused and 6 are replaced.)

15. ME 273-0046 dynatube fitting (Resistoflex, 24 each).

    NAA to remove, inspect, and reuse or replace. (For pricing, assume 18 are reused and 6 are replaced.)

16. ME 901-0067 rocket engine and ME 901-0189 nozzle extension (Rocketdyne, a Division of NAA, 12 of each).

    The engines have ablative thrust chambers and nozzle extensions which will be pyrolyzed and then soaked in sea water; their reuse is out of the question. The fine fluid passages in the injector system are critical and will be about the first point where corrosion attacks. There remains the possibility of salvaging the oxidizer and fuel valves, but valve seals may be susceptible to radiation damage and the cost of disassembly and inspection would tend to eliminate any savings. Although the engines are expensive, no prudent alternative to scrapping them is seen. All 12 engines and nozzle extensions must, therefore, be completely replaced.

Philosophy Behind Component Testing

Many of the components above (specifically, items 5, 6, 8, 9, 10, 11, and 13) are recommended for either replacement without inspection, or

retesting without disassembly and reuse based on testing only. The following considerations are the basis for these recommendations:

(1) Component development and qualification testing were designed to certify components for a single Apollo mission and do not provide test confidence for repeated mission use.

(2) Neither the component vendor nor NAA can be expected to certify the reliability with which a component will withstand a second mission, since there has been no test program on which to base such confidence.

(3) Any test program capable of developing sufficient confidence to certify components for repeated missions would be expected to cost more than the new components they would be designed to save.

(4) The process of machining open regulators, valves, etc. that are designed as welded systems, replacing all seals, reassembling, and retesting would be expected to cost almost as much as replacement with new components.

(5) The only alternative more economical than complete replacement of all such components is therefore careful component retesting at the component or subassembly panel level. Components passing this test would be declared reusable as a matter of engineering judgment only. Components failing such a test would be replaced either with new components or with components recovered from a second used CM. Since NAA has no test experience that certifies components for repeated usage, their reuse must reasonably be with the concurrence of and at the risk of NASA.

(6) The quantitative estimates made above of the proportion of components that might be recovered are considered the best feasible in the time available. These estimates might be improved somewhat by analysis of the repair and scrappage records of vendors, but the scope of such analysis is beyond the present contract. The only real way to determine how many components can be saved is to attempt an actual refurbishment.

POSTRECOVERY OPERATION REQUIREMENTS AND CONSTRAINTS

This analysis describes the precautions that should be taken during Apollo descent and post recovery periods to assure that the reaction control system is preserved for reuse to the maximum extent. The analysis was prepared with Spacecraft 012 in mind, but the conclusions reached seem equally applicable to Block II systems.

Spacecraft 012 has no automatic shutoff of RCS engine valves. The valves shut off when power is removed from the main d-c buses, but this may occur either before or after impact. It is important that the engine valves and the propellant isolation valves be shut to seal the interior of the RCS from sea water. It is, therefore, recommended that the Spacecraft 012 flight procedures be modified to require the following astronaut actions (if mission time permits), after the CM RCS purge system has been activated long enough to reduce the helium pressure (as monitored on the RCS indicator panel) to a low level (perhaps 100 psi), but before water impact:

1. Turn off the CM RCS Propellant Isolation Valve - A and B control switches.

2. Turn off any one of the following:

   (a) CM RCS Dump Switch

   (b) CM RCS Logic Switch

   (c) Main d-c power buses (if no longer needed).

In a normal mission there should be sufficient time in the parachute descent to accomplish the above actions. If the descent time should be too short in an unusual mission, the actions can be deleted but equipment damage will probably take place.

No procedure is now in use to preserve the exterior of the CM RCS from sea water degradation. One technique already discussed in the RCM study is to remove the lower heat shield and to hose down all accessible systems with fresh water. Such a technique, if done thoroughly, should provide satisfactory initial protection for the CM RCS. The propellant lines leading to the upper pair of CM RCS pitch engines may not be adequately rinsed in this operation, but they can be replaced if damaged; the engines are not considered to be reuseable in any event.

NAA Process Specification MA0210-0162 describes the postrecovery decontamination procedure required for the CM RCS. This procedure removes propellant from the interior of the CM RCS components down to a 300 parts-per-million level by repeated purging with methanol (fuel side), Freon MF (oxidizer side), and gaseous nitrogen. Failure of SPS tanks on SC 017 has shown that pure methanol is not a suitable material for use with titanium, and S&ID is now establishing a substitute for use in CM RCS decontamination. Otherwise, this procedure seems entirely appropriate for a CM RCS which is to be preserved for reuse. The procedure does require that the helium lines be crimped (1) close to Helium Bypass Valve A,

(2) close to Helium Bypass Valve B, and (3) close to the Helium Crossover Valve. All three valves are explosively actuated and would, therefore, have to be replaced before the system could be reused. The sections of helium lines containing the above crimps would also have to be replaced, but this creates no difficulty if they are close to the valves.

In review, the following actions are recommended to protect CM RCS components on Spacecraft 012:

(1) Shut off CM RCS engine valves and propellant isolation valves just before water impact.

(2) Remove lower heat shield and hose down the outside of the CM RCS thoroughly with fresh water.

(3) Decontaminate per specification MA 0210-0162 as presently planned.

Block II Spacecraft will have an automatic capability to shut off d-c power 13 seconds after CM RCS purge is initiated, but this is an optional feature that may not be used. Recommendation (1) is therefore still valid. CM RCS components will be packaged somewhat differently in Block II, but the changes should not affect recommendations (2) and (3) above. All three recommendations are, therefore, considered valid for both Block I and Block II systems.

EXPLORATORY TEST REQUIREMENTS

Exploratory tests may be defined as preceding and being in addition to component and subsystem acceptance tests; acceptance tests are those designed to show that the subsystem, as renovated, is ready for reuse. Review of the preceding sections shows that exploratory tests are not applicable to the CM RCS for the following reasons:

1. It is necessary to replace engines, nozzle extensions, and explosive valves in any event.

2. Any exploratory test of oxidizer and fuel tanks should certainly include disassembly and close examination of bladders and seals. Refurbishment, involving disassembly and replacement of bladders and seals, is believed more prudent, and should not cost a great deal more.

3. Helium tanks will be removed and reaccepted (given new proof and leak tests). No other tests on these tanks are considered necessary.

4. Acceptance tests of other components for reuse (valves, pressure regulators, etc.) will determine as much as can reasonably be learned of the functioning of a used component without disassembly. As outlined in the section on Renovation Requirements, disassemble of these welded systems is not considered cost-effective. The alternatives instead are acceptance based on functional tests and engineering judgement or replacement.

While exploratory tests are not proposed for the CM-RCS, the propulsion representative on the renovation team will analyze flight records and postflight subsystem appearance before recommending for a specific vehicle what components should be reused based only on acceptance tests.

XV. RELIABILITY

The considerations of reuse and test described herein are preliminary and intended for general guidance. They are based on the results of evaluations conducted to date by the responsible subsystems analysts and have been correlated with reliability considerations of mission success and crew safety.

REUSE CONSIDERATIONS

Vehicle reuse entails categorization of equipment into three general classifications: reuse without refurbishment, refurbish to varying degrees, or replace with a new item.

In assessing the impact of reuse on equipment reliability, consideration must be given to design criteria pertaining to time in environment, and equipment wearout, particularly of limited life items and moving or rotating components. These considerations affect any assumption concerning restoration to original status by refurbishment. In the current study, it has been assumed that (1) equipment is restored to its original status by refurbishment, and (2) where wearout is not a factor and no damage has been incurred, the probability of no failures for a successive mission of equal length is identical to the first mission, given that no prior failures have occurred and that design environments have not been exceeded. The latter assumption is based on the fact that electronic equipment generally possesses characteristics marked by the failure rate (bathtub) curve, illustrated in Figure 72.

For Apollo equipment not approaching wearout within the projected mission time, the "debugging" phase already has occurred. Therefore the useful life region, characterized by random failures, is the area considered applicable, and the same number of failures may be expected to occur in the time period t_1 to t_2 as in the equal time period t_2 to t_3. Restated, if R is the probability of no failures for time t_1 to t_2 and t_2 is reached with no failures, the same R would apply for time T_2 to T_3.

Electronic equipment of a static nature (e.g., transistors) are not typically subject to significant wearout within the mission times and equipment duty cycles associated with Apollo and RCM missions. Therefore, provided that an electronic component has not been subjected to physical damage, and provided that no failures have occurred during the first mission, this component should start the second mission with essentially the same inherent reliability it originally possessed. Other types of equipment are degraded to

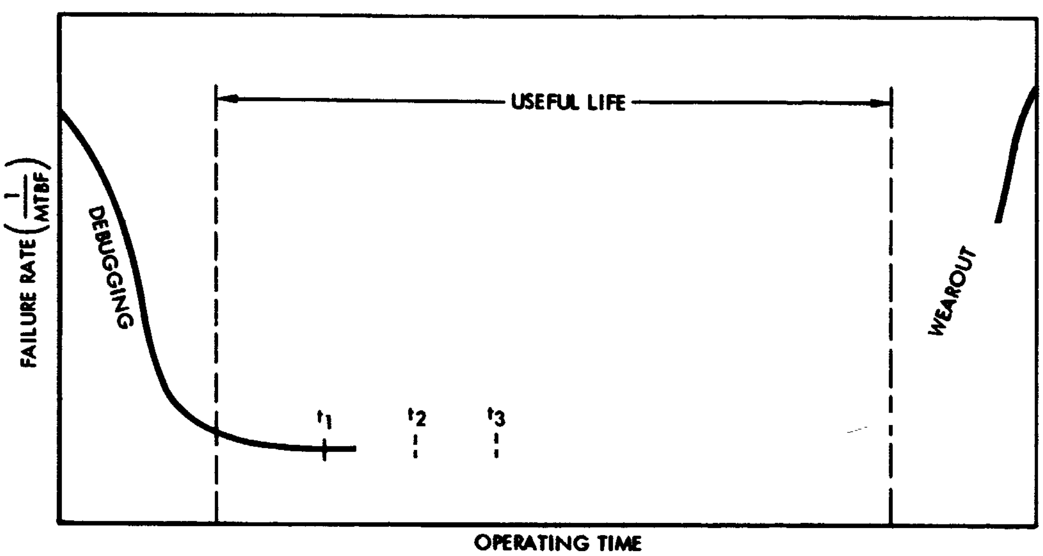


Figure 72. Failure Rate

varying degrees by operation and by the environments encountered during the mission. These are described in detail in the preceding subsystem sections.

TEST REQUIREMENTS

Testing requirements for equipment to be reused ("as is" or in a refurbished condition) depends to a great extent on criticality classifications, which are defined as follows:

Criticality I — A single failure or event which may expose the crew to environments beyond specified emergency limits.

Criticality II
1. A single failure or event which may cause an aborted mission.
2. Combination of any two failures and/or events which may cause loss of personnel. (Abort is mandatory after first unit failure following launch.)

Criticality III — All others.

The criticality of an item will vary, depending on whether it is intended for use in the RCM spacecraft or laboratory. The Criticality I and II item list for the RCM spacecraft will essentially be identical to the list for a basic Apollo spacecraft in an earth orbital mission but will be reduced extensively for laboratory application.

There appear to be few, if any, equipment items in the laboratory whose failure could cause loss of crew, with the exception of a catastrophic rupture of a pressure vessel. The remainder of possible failures to laboratory subsystems generally are related to experiment support. Their occurrence could reduce or eliminate the capability for conducting further experiments but not affect the status of the CSM equipment. While mission continuation might be pointless under these conditions, the failures are not classified as Criticality II, since termination of the mission for crew safety reasons alone is not required. Specific test requirements are defined in the subsystems section.

Qualification Tests

Basic Apollo equipment is being qualified for the LOR and other missions of similar duration. The RCM spacecraft mission is 14 days, which results in total mission time greater than 22 days for reused equipment. Refurbishment has been specified for rotating and limited life equipment, and other items subject to degradation. Under the study assumptions, these items are restored to original status insofar as reliability is concerned. However, pressure vessels installed in the command module should be qualified for the longer cumulative mission length if they are to be reused.

Certain critical equipment which may be used on the laboratory for the longer missions of up to 45 days is to be qualified for these longer missions. This particularly applies to Criticality I items, which would include pressure vessels and plumbing. In addition, requalification will be required for certain potential laboratory subsystems which have been relocated and will be subjected to more severe environmental operating conditions. In the case of other lower criticality equipment which has not been designed or qualified for longer missions, redesign or requalification is not considered essential. Failure of Criticality III equipment in the laboratory during the mission does not jeopardize crew safety.

As indicated in the reliability analysis section, failures can be tolerated in the laboratory which, if occurring in the CSM, would be cause for abort, For example, loss of two inverters in a CSM during an earth orbit would require abort, since failure of the one remaining inverter would eliminate essential inverter output. Loss of two inverters in the laboratory could be tolerated, since loss of the third would affect laboratory functions only.

Qualification by Similarity

In the case of certain types of equipment items (e.g., connectors, static electronic devices, etc.) wherein wearout normally would not be expected during RCM mission times, these items may be qualified for the longer mission time by similarity. If the supplier provides valid data regarding actual use of identical or similar equipment for comparable time periods without wearout, these equipments can be considered qualified, subject to the normal acceptance and all-systems test. Qualification by similarity have been evaluated on an individual basis by members of the project team.

Acceptance Tests

All equipment intended for reuse either with or without refurbishment generally must undergo the regular acceptance test specified for each item, including shock and vibration. In the case of items undergoing acceptance tests without refurbishment, the test results should be compared to those obtained prior to the original mission. If a specific performance parameter of the item undergoing test is within specification limits but indicates a decided drift towards these limits, when compared to previous test results, the item normally will be subject to rejection. Possible deviations will have to be evaluated on an individual basis by responsible project reliability and engineering specialists.

Exploratory Testing

In some instances, exploratory tests are to be conducted. These tests essentially consist of performance tests followed by disassembly and examination and are for the purpose of correlating postflight performance with actual degradation. Tests normally would be planned for the first recovered units in an attempt to arrive at valid conclusions affecting the degree of refurbishment of future units. Specific examples of possible exploratory testing are contained in the subsystem sections, particularly in the ECS/LSS section. The results of such tests are to be viewed with caution because of the limited sample size and the inherent risk of forming conclusions based on limited data; however, they may result on data which would provide partial justification for specifying refurbishment requirements.

Identification

If the RCM program is to be implemented, a system must be provided to identify components destined for reuse. The program ground rules have restricted the RCM spacecraft to earth orbital missions of 14 days. Items categorized as Criticality II or III when installed in the RCM spacecraft or laboratory might become Criticality I when applied to an Apollo lunar mission. Positive procedures will be required to prevent inadvertent installation of a

refurbished or already used component into a spacecraft programmed for a lunar mission. The cost of implementing adequate procedures must be considered in establishing a component cost point below which refurbishment will not be economically feasible.

RCM AVAILABILITY ASPECTS

Previous studies conducted at S&ID (e.g., Manned Mars and/or Venus Flyby Vehicle Systems Study, SID 65-761-5) have concluded that maintenance is required to assure adequate probabilities of mission success and crew safety for long space missions. While the baseline mission for the Block II renovated spacecraft is only 14 days, mission of up to 45 days are planned for the RCM laboratory. The fully dependent laboratory version will have a minimum of equipment installed: largely interface with the su orting command module electrical power, environmental control, and communications. The independent laboratory will be essentially a fully equipped spacecraft with the attendent reliability problems. It appears that maintenance is more applicable to the independent laboratory than to the other configurations; however, most candidate subsystems have not been designed for maintainability and, in general, maintenance will be severly limited without redesign.

The Availability Concept

System availability is defined as that percentage or fraction of the total desired operating time that an equipment is in commission (ready for use)[1]. Numerically this is defined as

$$\text{Availability (A)} = \frac{\text{MTBF}}{\text{MTBF} + \text{MTTR}} \tag{1}$$

where

MTBF = Mean Time Between Failures - (MTBF is applicable to the region in which random failures occur and does not apply to the wearout region. Where wearout is expected within the mission time, provisions in the form of standby redundancy or spares also must be incorporated.)

MTTR = Mean Time to Repair - (Including trouble isolation time.)

[1] Availability is not a new measure of reliability. Reliability can only be improved by a successful maintenance action. Thus the probability of an equipment being in the commission state is improved by the successful maintenance action.

Availability, like reliability, is a probability; it is expressed as the probability of an item of equipment being operable at any given point in time.

Since availability is a function of MTTR, it is subject to certain constraints related to the ability of the crew to accomplish the required maintenance actions and other constraints related to the requirements of the spacecraft and mission. For example, if a certain equipment is required to perform a function and the function can be delayed without jeopardizing crew safety or mission success, the reliability requirement may be relaxed providing the equipment is repairable. Some functions (e. g., life support) obviously cannot be interrupted for more than extremely short time periods. A detailed failure mode and effects analysis must be accomplished before a positive determination relevant to specific functional availability requirements and constraints can be made.

It follows that sufficient spares must be provided to support maintenance actions planned for those maintainable subsystems wherein inherent reliability does not meet requirements.

The probability of mission availability (A_M), assuming sufficient spares for a single unit, is:

$$A_M = e^{-\lambda T e^{-ut}} \tag{2}$$

where

λ = equipment failure rate, $\frac{1}{\text{MTBF}}$

T = mission time

u = repair rate = $\frac{1}{\text{MTTR}}$

t = maintenance time constraint (MTC)

The average number of repairs (r) expected may be estimated by

$$R = \frac{\text{Mission Time}}{\text{MTBF}} \tag{3}$$

The overall system availability is

$$A_s = \prod_{i=1}^{i} A_{M_i} \tag{4}$$

RCM Application

Application of the availability concept to the renovated command module is limited by the following study ground rules and mission constraints:

1. No maintenance can be accomplished by the crew during periods of acceleration.

2. Basic subsystems to be installed in the laboratory or spacecraft will not be subjected to major redesign and attendent additional development.

3. Maintenance will not be accomplished through extravehicular activity.

It is noted that the Block I Apollo design initially incorporated maintenance provisions for the lunar mission and provided for approximately 150 pounds of spares, largely electronics. In-flight maintenance capability for the electronic subsystems was removed from the Block II configuration at the direction of NASA: therefore, improvements of the system availability for the RCM spacecraft (Block II) generally does not appear feasible. Maintenance may be feasible without extensive modification for certain RCM laboratory subsystems whose performance would otherwise be marginal, particularly on the case of electronic modules, which may be removed and replaced. However, this will require fault isolation capability. Since failure of components located within the laboratory does not have a degrading effect on crew safety, an extensive equipment modification program does not appear warranted.

Spares

Active redundancy is frequently used in situations where the reliability of a single unit is inadequate. Active redundancy has certain limitations, particularly in the case of life-limited items, since the redundant unit also is accumulating operating time. Standby redundancy is more effective for longer mission times, but adds the complexity of a switching mechanism. Spares may be regarded as a form of standby redundancy where man is the switching mechanism. To determine the reliability obtained by the use of spares, the Poisson distribution as listed below is applied. In this application the spare is assumed to have the same failure rate as the unit it is intended to replace and the switching mechanism (man) is regarded as unity.

$$R = e^{-\lambda t} + (\lambda t)e^{-\lambda t} + \frac{(\lambda t)^2 e^{-\lambda t}}{2!} + \frac{(\lambda t)^n e^{-\lambda t}}{n!} \qquad (5)$$

where

λ = failure rate of units

t = system duty cycle

n = number of spares

XVI. RCM LABORATORY SUBSYSTEM BUILDING BLOCKS

A limited and incremental spectrum of subsystem functional capability has been provided through the generation of a RCM Laboratory subsystem building block shopping list. The underlying theme here is a combination of economics and expediency. By selecting the subsystems and components for this list from those space-rated items which are presently available, development and qualification are minimized and in most cases eliminated. Apollo CM and SM equipment comprise most of the listed items with subsystems and components from LM, Gemini and the Agena Gemini Target Vehicle also present. Equipment from other space programs (such as Mercury, OGO, etc.) was also considered.

This approach to satisfying RCM Laboratory mission objectives appears sound as long as overall system weight is not critical and there is a large degree of flexibility in the mission objectives. The system configuration established by resort to the shopping list with a mission to establish criteria, could dictate certain changes to the mission (contracting or expanding the objectives) in order to provide a higher cost effectiveness. This approach is necessitated by the discontinuity in the available subsystem functional spectrum.

The process for integrating the shopping list subsystems into an RCM Laboratory system is shown in Figure 73. The overall subsystem integration procedure requires an iterative facet which accounts for the subsystem interfaces with each other. Once the subsystems are defined by resort to the shopping list, with the mission-dependent requirements as criteria, interface requirements (such as power, temperature control, environment control, structural, etc.) are determined and the subsystem requirements are adjusted accordingly.

The subsystem selection process is repeated, and after subsystem definition, the interface requirements iterative action is repeated. This cycling continues until no change in the defined subsystems results from an interface iteration. At this point, subsystems are integrated into the laboratory and the required delta RCM Laboratory configuration results.

The following portions of this section discuss and identify the subsystem building blocks. Where critical, integration constraints are detailed and delineated.

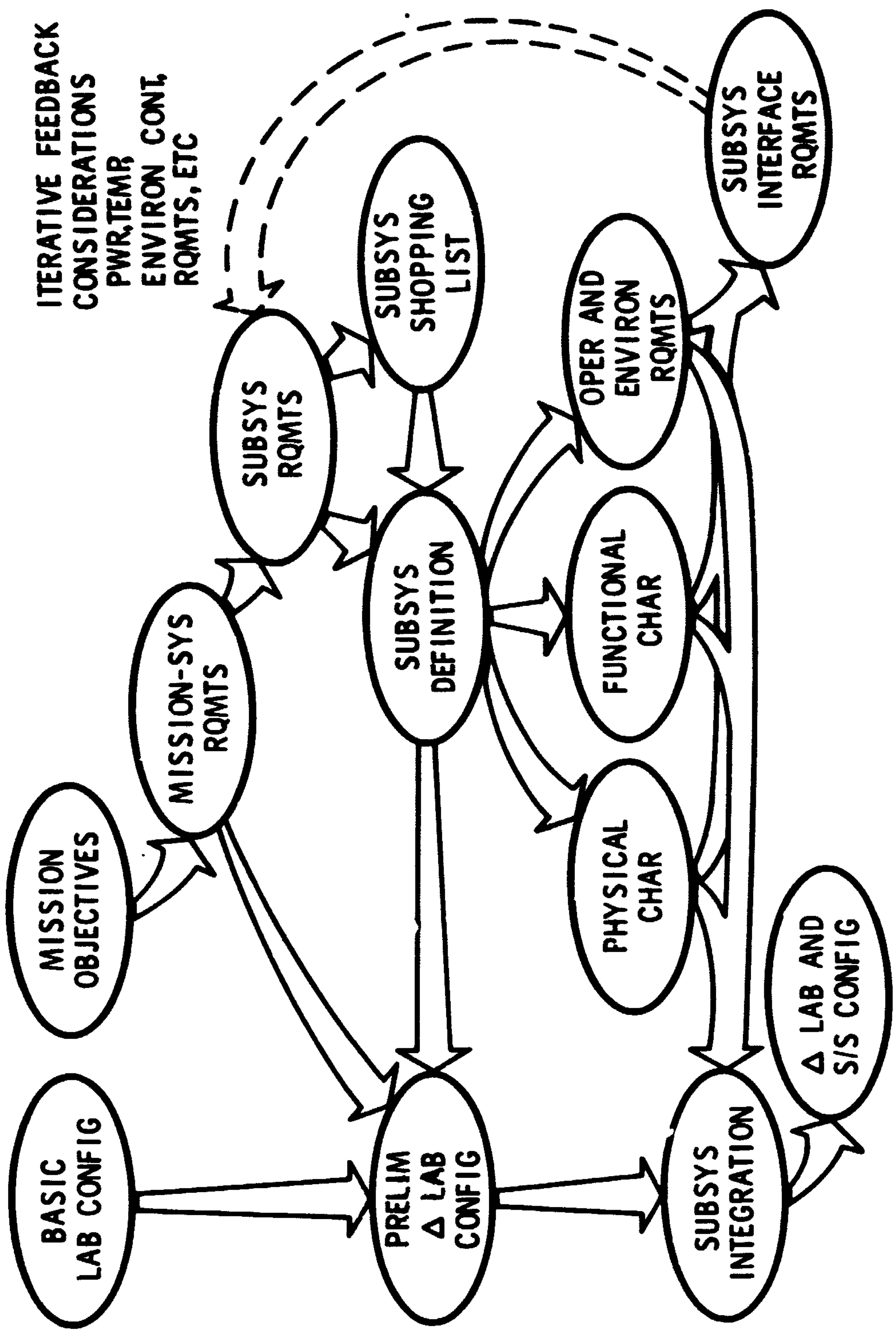


Figure 73. Subsystem Integration

ENVIRONMENTAL CONTROL SYSTEM/LIFE SUPPORT SYSTEM (ECS/LSS)

The only possible ECS/LSS available, other than Apollo's, for use on the RCM Laboratory would be equipment used on the Mercury and Gemini programs.

Characteristics of Other Applicable ECS/LSS Subsystems and Components

The Environment Control System for the Mercury, Gemini, and Apollo spacecraft are very much alike in principle and probably have many similar components. All three spacecraft maintain a pure oxygen atmosphere at 5.1 psia. However, the Mercury and Gemini are essentially a separate b . suit circuit with a cabin backup while the Apollo system is designed with emphasis on maintaining a shirtsleeve cabin environment with a suit circuit backup.

The design of the systems for the three spacecraft was a progression from one to the other. However, as the second and third crew members plus higher electrical cooling loads were added, the Mercury design concepts had to be modified to some extent. Generally, many of the Mercury and Gemini ECS design concepts have been carried through to the Apollo, but the added heat loads, installation limitations, efforts to reduce weight, and the intrusion of a limited number of completely new concepts nevertheless made it necessary to successively modify the Mercury and Gemini hardware. It is felt that the design constraints of Mercury and Gemini equipment would not lend to renovation for use on the RCM Laboratories.

Table 36 outlines the characteristics of the three systems. Examination of Table 36 will show that the Mercury system would be totally inadequate for RCM or RCM Laboratory application. The one-man, short-time oxygen flow and heat capacity would not lend itself to a thirty-day mission.

The use of a Gemini system as a building block or in conjunction with an Apollo system is also not feasible for two main reasons: 1) incompatibility of coolants, and 2) incompatibility of system capacities. The Gemini uses coolanol for a coolant while the Apollo uses water-glycol. The specific heat and viscosity of the two fluids differ sufficiently to make equipment designed for one incompatible with the other for both hydraulic and heat-transfer reasons. The Gemini has a high ΔP coolant pump to accommodate a series flow circuit. The Gemini radiator is an integral part of the structure and cannot be mounted separately for an RCM Laboratory as the Apollo radiator can.

The Gemini heat-transfer surfaces and oxygen-flow capacities are designed for two men and not three as required. The use of two two-man

Table 36. Functional Characteristics of Candidate ECS/LSS

	Mercury	Gemini	Apollo
1. Mission duration	72 hours	14 days	14 days
2. Size of crew	1	2	3
3. Capsule internal pressure (nominal), psia	5	5	5
4. Capsule internal temperature (nominal), F	80	60-80	70-80
5. Capsule atmosphere	Oxygen	Oxygen	Oxygen
6. Capsule internal volume, ft^3	50	73	233
7. Capsule ventilation rate (nominal), cfm	22	88	300
8. Crew heat load (nominal), Btu/hr	500	1000	1400
9. Crew oxygen consumption, lb/day	2.0	4.0	6.0
10. Crew CO_2 production (nominal), lb/day	2.25	4.50	6.75
11. Suit ventilation rate, cfm	11.4	23.0	35
12. Suit compressor rise, in. H_2O	10	11	10
13. Emergency suit pressure, psia	4.0	3.5	3.5 to 4.0
14. Emergency oxygen flow rate, lb/min	0.05	0.1	0.67
15. Coolant used	Water	Coolanol	Water-Glycol
16. Coolant inlet temperature to suit heat exchanger, F	40	40	45
17. Coolant flow rate, lb/hr	0.5 to 5	170	200
18. Coolant pump pressure rise, psi	No Pump*	200	36
19. Electrical load (nominal), watts	N.A.	500	1300
20. Electrical load (maximum), watts	N.A.	1500	2300

*Water moved by O_2 pressure
NA: Not available

systems plus component modifications to obtain a three-man system would probably not be economically feasible from a retest and requalification standpoint.

Proposed Environment Control and Life Support Systems

The recommended ECS/LSS for the RCM Laboratory would be basically renovated equipment from the Apollo program. The systems for the basic (dependent) and the independent laboratory are outlined below.

ECS/LSS For Basic or Dependent Laboratory

The ECS for the dependent laboratory will consist of a blower and duct to promote sufficient exchange of atmosphere between the CM and laboratory to control humidity, carbon dioxide and oxygen concentration, and temperature in the laboratory by making use of the CM/ECS/LSS.

Data for this study were taken from Reference 12 which contains a lengthly section on the conditioning of the LM by use of the Apollo ECS/LSS. This data is directly applicable to the dependent laboratory requirements.

Reference 12 contained parametric studies on the effect of compartment atmosphere interchange upon the control of carbon dioxide, humidity, oxygen concentration, and temperature. The following tabulation is taken from this data and presents the minimum atmospheric-flow requirements for each of the parameters if three men were in the laboratory:

Parameter	Minimum Atmospheric Flow (cfm)
Humidity	120
CO_2	60
Temperature	425
O_2	40

The atmospheric flow rate for humidity control is based upon a dew-point differential of 3 F between the two compartments. The oxygen flow rate is based upon a 2-mm Hg difference in oxygen partial pressure, and the interchange rate for carbon dioxide is based upon a carbon dioxide partial pressure difference of 0.5 mm Hg and nominal generation of carbon dioxide. The use of simple compartment atmosphere interchange for thermal control of both compartments is based upon the temperature limits of 75 ± 5 F. The laboratory can then be at 80 F while the CM can be at 70 F. The 10 F temperature difference imposes a severe penalty in terms of fan power to

accomplish all the thermal conditioning by means of this approach. From the tabulation, it is clear that an interchange flow rate of 120 cfm is required to maintain a 3 F dew-point differential. At this flow rate, carbon dioxide and oxygen partial pressure control requirements are amply satisfied.

The RCM Laboratory will never contain more than two men with one man always required in the CM. Figure 74 was constructed to show the effect of two men in the RCM Laboratory. These data assume no heat loss in the laboratory. Some heat loss in the RCM Laboratory will be beneficial because it will reduce the temperature difference between the two compartments.

The fan selected for intercompartment circulation is the postlanding ventilation fan without modification. The characteristics of this fan are as follows:

Flow	180 cfm
Pressure rise	0.2 in. H_2O
Weight	4.5 lb
Power	13 watts

The use of this fan is recommended because of the relatively low power requirement associated with the high flow rate. Although the flow rate is 50 percent higher than necessary to control the dew-point in the two compartments to less than a 3 F differential, the extra circulation is recommended because it will reduce the atmosphere differences between compartments and it will cause the flow distribution within the laboratory to be more uniform. The 180-cfm flow plotted in Figure 74 shows CM temperature of approximately 65 F.

The fan circulates the CM atmosphere into the laboratory by use of a 5-inch diameter duct located in the CM laboratory tunnel. The duct could be a rigid tube fastened to the side of the tunnel after mating pressurization of both modules. The exit of the duct located in the laboratory should exhaust the circulated gas along the wall of the laboratory to avoid short-circuiting the gas flow. The tangential injection also would enhance the mixing and purging of the laboratory atmosphere.

Table 37 shows two additional dependent laboratory ECS/LSS deltas.

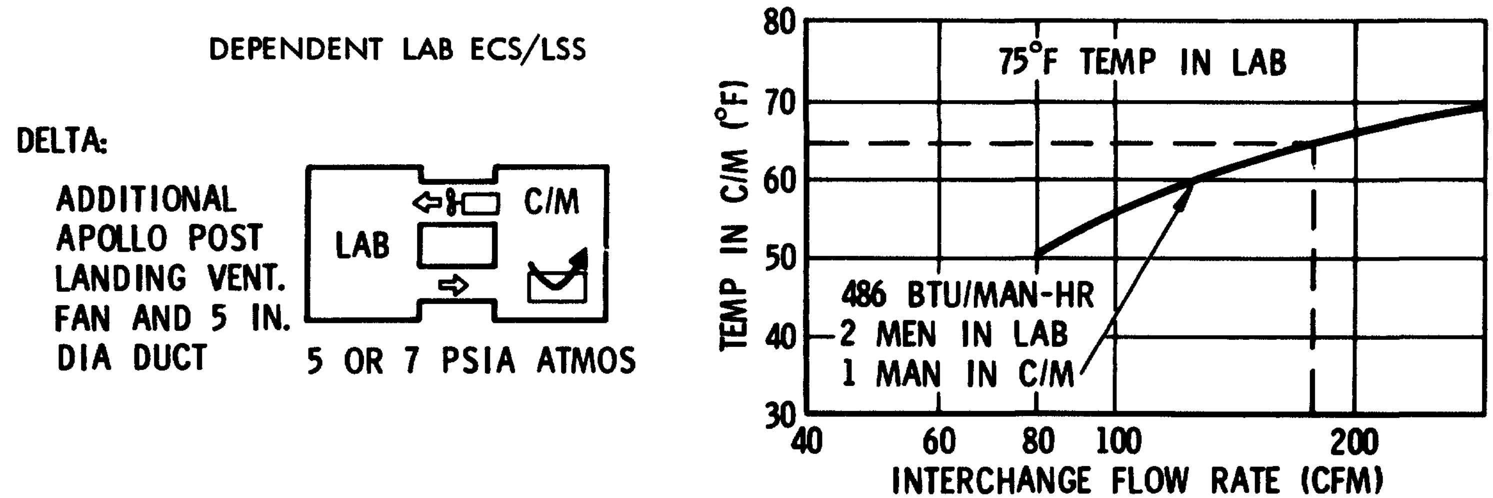


Figure 74. RCM Laboratory ECLSS

Table 37. Dependent Laboratory Configurations

Configuration	Limitations and Conditions ↓ Mission Duration →	Deltas in RCM Laboratory 15 Days	30 Days	45 Days
Dependent laboratory	1. Low power load in laboratory 2. All power from CM 3. Passive cooling of equipment in laboratory	PLV blower + duct		
Δ1	1. Intermediate power load in laboratory 2. All power and water from CM 3. Coordinate operation of CM and laboratory pressure control systems.	1. PLV blower + duct → 2. Coolant loop and heat rejection capability → a. water boiler b. radiator c. both water boiler and radiator	1. → 2. → 3. O_2 and/or N_2 supply and control system →	1. 2. 3. 4. Additional O_2 and/or N_2 storage capacity
Δ2	1. High heat loads in laboratory 2. Power and water source in laboratory 3. Coordinate operation of CM and laboratory pressure control systems	1. PLV blower + duct → 2. Coolant loop with water boiler and radiators → 3. Cabin temperature control system → 4. Water management system →	1. → 2. → 3. → 4. → 5. O_2 and/or N_2 supply and control system →	1. 2. 3. 4. 5. 6. Additional O_2 and/or N_2 storage capacity

ECS/LSS For Independent RCM Laboratory

Table 38 summarizes the independent RCM Laboratory ECS/LSS configuration plus deltas required to extend mission lengths from 15 to 30 to 45 days.

15-Day Mission. A fifteen-day mission would require no more than a standard renovated Block II ECS/LSS.

30-Day Mission. A thirty-day mission would require a standard renovated Block II ECS/LSS plus an optional one-gas to two-gas atmosphere control and supply system plus an extra fifteen-day supply of LiOH canisters that can be transferred from the CM to the laboratory when needed. An optional one-gas or two-gas atmosphere control and supply system can be accomplished by the addition of a nitrogen low-pressure supply system and a oxygen partial-pressure sensor and control system. The nitrogen and oxygen low-pressure supply circuits would be interconnected so the both could be controlled by the present Block II cabin total-pressure regulator. When only an axygen system is desired the nitrogen is valved off and the system operates as a standard Block II system. When it is desired to operate as a two-gas system, the system is valved so that the nitrogen flow is controlled by total-pressure regulator and the oxygen flow is controlled by the oxygen partial-pressure regulator.

Table 38. Independent Laboratory Configurations For Various Mission Lengths

Mission Duration	Basic Unit	Deltas
15 days	Block II ECS/LSS including LiOH absorpiton unit	None
30 days	Block II ECS/LSS including LiOH absorption unit	Extra LiOH in CM Optional one or two-gas atmosphere control and supply system.
45 days	Block II ECS/LSS including LiOH absorption unit for suited operation only.	Independent molecular-sieve unit Optional one or two-gas atmosphere control and supply system

These optional atmosphere systems will permit the use of either a one-gas or two-gas CM. Also, the approach incorporates versatility of operation required by the fact it permits making use of the advantages of either a one-gas or a two-gas system as desired.

45-Day Mission. A forty-five day mission configuration would require the same standard renovated Block II ECS/LSS system and optional atmosphere control and supply systems for the 30-day mission plus a molecular-sieve unit for carbon dioxide removal. In order to obviate any redesign of the renovated ECU, it is required that the molecular-sieve unit be a complete and integral unit with its own blowers and ducting system. The existing LiOH absorption unit will be used only during suited operation. When the molecular-sieve unit is in operation, the LiOH is not used unless someone is in a suit.

Integration Constraints

When integrating the Apollo ECS/LSS equipment into the RCM independent laboratory, it is assumed that all equipment items originally mounted inside the CM will again be mounted inside the RCM and that only items originally part of the SM will be mounted outside the RCM.

Attention is called to the following major operational constraints that must be considered when adapting the Apollo equipment for use on the independent RCM laboratory.

Leakage and Operation

When the CM and the independent RCM Laboratory are docked and each has its own pressure control system, considerable thought must be given to coordinated operation of the two systems. For example, if the allowable tolerances (±0.2 psia) of the total-pressure regulators were such that one was controlling at 4.8 psia and one at 5.2 psia, the regulator admitting gas at 4.8 psia would supply makeup for the leakage of both vehicles, thus depleting the system of one of the spacecrafts.

In the case of a two-gas atmosphere control system, the oxygen partial-pressure control system of one spacecraft might supply all of the oxygen makeup if its tolerance was such that its setting was below that of the other spacecraft's system. This problem could most easily be solved by keeping the interconnecting hatch closed as much as possible. This approach would call for no system changes or redesign.

Another alternative would be to provide a means to shut off one system before it becomes depleted and let the other take over. This approach would require close monitoring of expendables but only minimum design changes.

Heat-Load Ranges

In order to maintain a shirtsleeve environment in the laboratory, the net heat load (summation of losses and gains) must be within the capability of the Apollo ECS. The net capability is defined by the capability of the cabin heat exchanger which has a cooling capacity of 760 Btu/hr and a heating capacity of 236 Btu/hr.

A major factor regarding the heat load is the vehicle heat transfer in and out of the conditioned volume. This heat load must be controlled within a range such that gains in this area do not exceed system cooling capability or allow condensation to form on walls. Using Apollo as a guideline, environmental load should be negative and range between -200 to -1000 Btu/hr.

Radiator Subsystem

The Apollo ECS radiator system was designed for a specific set of conditions of heat load, water-boiling supplemental heating, and spacecraft orientation. In the Apollo system, when heat loads exceed radiator capability, supplemental cooling is accomplished by the water-glycol evaporator. When the rejection capacity of the radiators exceeds the heat load, supplemental heaters add energy to the coolant to prevent the coolant from freezing in the flowing radiator tubes. It is felt that the possibility of freezing the coolant in the radiator panels points out the major problem with respect to direct application of Apollo systems to laboratory utilization. This is based on the assumption that, for the extended mission, the power load will be low for extended periods of time. If this assumption is correct, the radiator system as presently configured will consume a large amount of electrical power.

Table 39 shows the general performance characteristics under various environmental conditions of the Block II radiator subsystem. As can be seen, considering no sun on radiator as the worst-case environment, the minimum load the radiator can receive without supplemental heating is estimated at 5600 Btu/hr. To maintain this heat load to the radiator system in the case of Apollo, a gross fuel cell power load would be on the order of 2000 watts. In addition, the 4415 Btu/hr load with no sun on radiators requires a 290 watt heat load for a three fuel cell minimum load of 1689 watts. Table 39 also shows that 2 panels will reject greater than 9275 Btu/hr with no sun on radiators, which could be translated into a gross power loads greater than 3317 watts.

Table 39. Block II Radiator Subsystem Performance Characteristics

Heat Load (Btu/hr)	Environment	Bypass (%)	Water Boiling (lb/hr)	Heat Input Watts/Btu/hr
Two Panels				
9275	No sun on radiator	20	--	--
5600*	No sun on radiator	30	--	--
4415	No sun on radiator	65	00	290/1000
6880	Thermal cycle	30	--	--
6880	Sun on one panel	10	--	--
5500 ± 400	X axis parallel to lunar surface	--	0.85	--
5600*	Earth orbit	--	--	--
One Panel				
6750	Thermal cycle	--	2.0	--
4415	Thermal cycle	--	0.5	--
2920	No sun on radiator	70	--	--

*Estimated minimum load without supplemental heating.

The Block II radiation is presently configured to be operated with one panel. In this situation, the nonoperating panel is allowed to freeze with no provisions for activation other than spacecraft orientation. Table 39 under one panel shows the system performance for a single panel. Here, it can be seen, that the minimum load required without supplemental heating is reduced to 2920, with no sun on radiators. For the Apollo mission, this heat load can be translated into two fuel cell minimum power of 1126 watts.

As mentioned above, if power loads are low for extended periods of time, Table 39 shows that the heater power required by the radiator system can be reduced by utilizing single panel operation. With single panel

operation, a heater will be required on the radiator panel to reactivate the frozen panel when heat load demands exceed single panel capability.

If reduction to single panel operation would still require large amounts of cryogenic fluid for heater operation, an additional modification to the existing panel could be accomplished without significant added effort. This modification to the panel would be to remove the series tube section and the selective stagnation section only.

Figure 75 shows critical integration constraints and indicates the Block II ECS gross capabilities and limitations for the design mission for RCM Laboratories. The performance of the radiator subsystem is bounded by water boiling for the maximum heat load and heater power for minimum heat load. In this respect, Figure 74 shows the average range of heat loads that can be accommodated by the radiator subsystem without use of water boiling as supplemental cooling or heater power to prevent the radiator panels from freezing. Missions should, therefore, be planned on a preliminary basis within the range of heat load that would not require expendables, in either the ECS or EPS systems. For example, in the low inclination earth orbit, mission total heat loads to the radiator system should be maintained between 3500 and 4700 Btu/hr., and in no case should the total load drop below 1900 Btu/hr for an extended length of time. Total heat load could exceed 4700 Btu/hr providing there is sufficient water for supplemental cooling. The same could be said for the synchronous equatorial earth orbit mission for the ranges and heat limitations shown.

It should be noted here that the combination of rejection capability and radiator rejection for lunar orbit causes the rejection capability bar to reverse. That is, the minimum capability established by dark side operation exceeds the maximum capability on the light side. This indicates clearly that, for mission planning purposes, heat loads should be high on the dark side and low on the light side. In the case of Apollo Block II, radiator heat loads are approximately constant through lunar orbit with water boiling on light side not to exceed net water production.

In the case of extended mission duration, the power loads and heat loads to the radiator are low, which would require excessive power consumption by radiator subsystem heaters. In order to avoid this situation, the radiator subsystem could be operated with only one of the two panels in operation. Figure 75 also shows the estimated minimum heat load without heater operation. The minimum heat-load capability was based on flow-rate limitations and was estimated at 60 percent of minimum load of two-panel system.

It also should be noted that, if a single-panel operation is considered, an additional startup heater on the deactivated panel will be required to thaw panel prior to high heat-load periods. With a heater incorporated in the

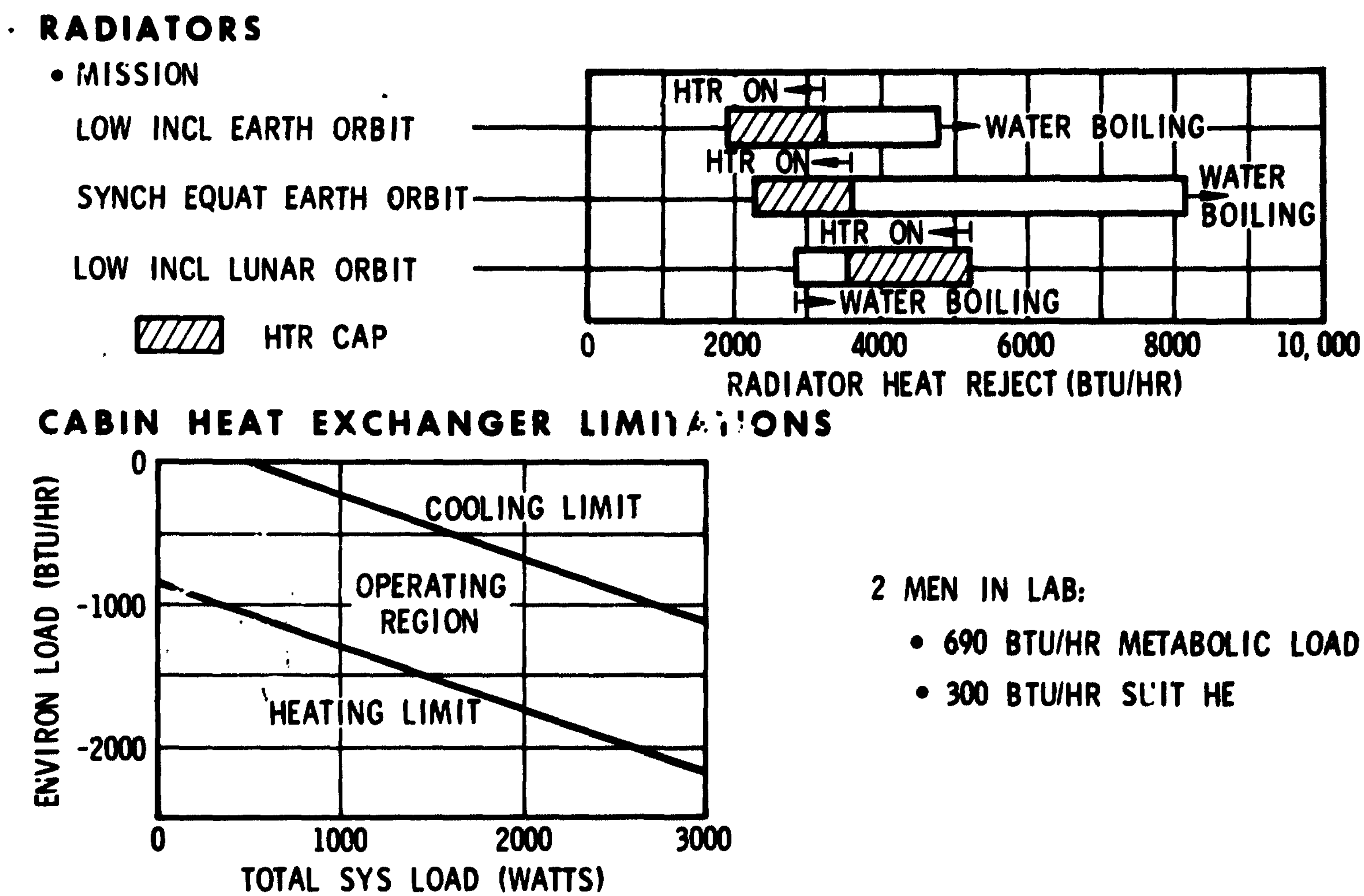


Figure 75: ECLSS Integration Constraints

panel design, an addition panel could be installed and activated in the event of meteroid puncture.

Several additional comments should be made with respect to ulilization of the Block II ECS radiator subsystem for the independent laboratory. First, the primary loop of the radiator subsystem can be utilized with either a Block I or Block II command module ECS hardware. Second, if the radiator panel requirements for meteoroid puncture are greater than Apollo lunar mission requirements, additional analytical efforts are required. These efforts would be directed toward utilization of both primary and secondary loops in the Apollo Block II system in the same fashion as Apollo after one radiator panel has been punctured. The analysis would involve the rejection capability of the system versus the functions or experiments that could be accomplished at a limited rejection capability.

It should be noted here that. if dual loop operation is considered, all Block I ECS systems will have to be modified to incorporate secondary loop components and Block II coldplates. If, on the other hand, dual loop operation will not satisfy requirements, additional analytical and test efforts will be required to define and verify redesign.

Figure 75 also shows the cabin heat exchanger limitations. The cabin heat exchanger ties together the environmental loads and the total system power loads. Without exceeding the exchanger capability limits, the system power loads may swing from 400 to 2800 watts if the environmental load is a loss at 1000 Btu/hr. If the environmental load were zero (laboratory thermally isolated) system load range would be only 0 to 500 watts.

Physical Characteristics

Power Requirements

Power requirements for the ECS/LSS are tabulated in Table 40. The power requirements for the PLV fans and valves occur only after touchdown. The 45.0 watts for potable water occur during heating water for food preparation. No attempt has been made to add up the various column to reach a total because not all items will be operating at the same instant. A more detailed definition of the various mission profiles is needed to reach an average or transient total.

Weight

The ECS/LSS in the CM weighs 536 pounds wet and 512 pounds dry. The ECS/LSS equipment in the SM weighs 160 pounds wet and 144 pounds dry.

Volume and Installation Requirements

The volume and installation details for the various sections of the ECS/LSS are covered by the NAA Apollo installation drawings listed in Table 41. Most of the equipment is located in the lower left-hand equipment bay on the Apollo.

Table 40. ECS Power Requirements (Watts)

Item	Block I				Block II			
	DC		AC		DC		AC	
	High	Low	High	Low	High	Low	High	Low
Suit Compressor (one compressor operating)			232.0	85.0			232.0	85.0
Glycol pump (one pump operating)			36.0	36.0			50.0	50.0
Cabin fan (two fans operating)			60.0	38.0			60.0	38.0
Radiator control valve (two valves operating)			14.4				14.4	
Glycol temperature control, sensor, and control valve	0.001		9.7	2.5	0.001		9.7	2.5
Cabin temperature control, sensor, anticipator, and control valve	0.002		9.7	2.5	0.002		9.7	2.5
Backpressure control, sensor, and control valve	1.28		9.7	2.5	1.28		9.7	2.5
Wetness control, sensors, and valve	0.28	0.28	5.0	5.0		0.28	8.0	3.0
Power supply	10.5	10.5			10.5	10.5		
Miscellaneous valves, sensors, controllers, transducers	24.8				45.6			
PLV fans and valves	70.0	10.0			111.55			
Radiator heaters and controllers			14.4		976.0	5.6	13.0	
Potable water	45.0				45.0			

Table 41. Installation Drawings

Item	Block I (12)	Block II (101)
ECU	V16-613001	V36-613500
Cabin recirculation unit	V16-610003	V36-613800
Coldplate network	V16-610004	3V36-610001
Water systems	V16-610005	V36-610201 V36-610500
Oxygen system	V16-613001	V36-611500
Suit hose connection assembly	V16-610022	V36-613800
Waste management system	V16-610006	V36-612501
Radiators		V37-32201

ELECTRIC POWER

A specific power system for the independent RCM Laboratory cannot be designed because of the lack of definite mission requirements. However, the objective of utilizing as much Apollo equipment as possible does allow outlining the capability of the power system in terms of potential missions. This capability has been looked at from two viewpoints. The first centers around the physical problems of mounting a complete Apollo Block II service module power system on the cruciform structure. The second is to define the physical and functional properties of the EPS components and provide a guide for building up a complete system when a definite mission has been established.

The baseline design configuration for the purposes of structural consideration is based on the existing CSM power system. Block II service module equipment is used because the original equipment is not recoverable and new components must be used. The configuration using Block I equipment in the RCM Laboratory and Block II equipment on the cruciform can occur. This will not create any serious problems because, even though the components are mechanically different, they have similar electrical interfaces.

Three fuel cells and two sets of cryogenic tanks have been selected to demonstrate the feasibility of locating this system on the RCM Laboratory. This type of system is arbitrarily chosen because it utilizes Apollo equipment to the fullest extent, and it is probably the most complicated and difficult to install of the several potential systems. The system actually used must be selected on the basis of detail mission requirements; to keep costs down, as much existing hardware as possible must be used.

In order to utilize the maximum amount of the Apollo system as possible, each component's capability within the system must be defined. The basic power system used in the CSM is shown in the block diagram of Figure 76. The relationship of the various power sources to one another is shown and the most likely points at which a substitution or change may be made can be ascertained. For instance, substitution of larger capacity batteries for batteries A and C, removal of all other power sources, and disabling all unused control circuits would create a workable primary battery system.

Dependent Power Source

Selection of the main power source is one of the most serious problems. Three possible types have been studied for the RCM Laboratory. The first is the simplest and consists of obtaining power from the CSM. A study (Reference 13) has been made for the AAP program on the power available for an external laboratory. The results show that it is feasible to consider delivering as much as 1200 watts from an AAP Apollo CSM. Also, there is

APOLLO EPS FUNCTIONS

- REDUNDANT BUSES
- BUS MONITORING
- CKT PROTECTION
- FUEL CELL AND BATTERY CHARGE CONTROL
- 3 ϕ AC INVERTERS (1250 VA @)

FLEXIBILITY

- INHERENT FUNCTIONAL BLDG BLKS
- FAILURE MODE OPERATION

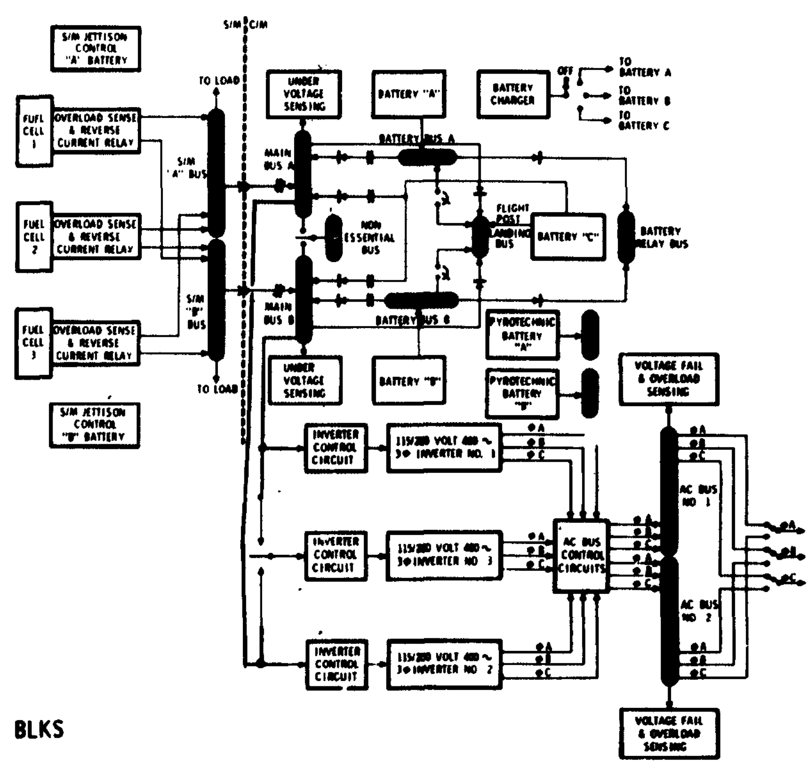


Figure 76. Laboratory EPS Distribution Control and Conditioning

at present in Block II Apollo a LEM power feed-through in the docking tunnel designed to deliver power at 200 watts. The primary constraint for this system is the cryogenic storage capability of the CSM. The AAP CSM utilizes five pairs of cryogenic tanks. If a Block II CSM is used, a 200-watt level would probably be the maximum power level that could be maintained for two weeks.

Battery Power Source

Several batteries have been designed and qualified in the Gemini and LEM programs in addition to the Apollo batteries. They are all of the silver-zinc type and therefore have a very limited charge/discharge cycle life. These batteries are listed in Table 42 along with their pertinent characteristics. Exclusive use of the large batteries (300 amp-hr) would eliminate any requirement for the Apollo battery charger because of its small capacity. This limits the use of the large batteries to a primary battery system. The most likely use for the smaller batteries would be for supplying power for cyclic peak-load conditions. Under these circumstances, use of the battery charger would be in order.

Fuel Cell Power Source

Fuel cells are the primary source of power on the Apollo CSM and as such are strong candidates for a RCM Laboratory power source. The EPS components available for reuse have been designed to protect and control fuel cells, thus eliminating many problems that would occur if another type of source had been used. However, a fuel cell is an active device which requires a very complex system to support it. Many subsystems are effected in these support requirements and must be investigated to insure successful operation. A cryogenic oxygen and hydrogen storage system with sufficient capacity to supply the necessary reactants must be installed. A radiator system is also needed to dissipate the waste heat generated by the fuel cell. These two subsystems themselves generate a host of problems that must be investigated and solved; such as limiting the cryogenic system heat leak to prevent venting the reactants overboard and to prevent freezing of the water-glycol in the radiators. A solution to these problems cannot be established until a definite mission has been defined and the effects of time, orbit, and objectives evaluated. The pertinent fuel cell and supporting subsystem performance characteristics needed to evaluate their applicability and to provide a preliminary design estimate are presented as an alternate to a detailed fuel cell system design.

Batteries, fuel cells, and combinations of both are characterized and limited by power, energy, transient capability, weight of reactants required, volume of power system including reactant tank system, temperature, radiator heat rejection capability, reliability, cost, and availability. Although

Table 42. Alternate Batteries

Battery	Program	Capacity (a-hr)	Voltage (vdc)	Type	Part No.	Manufacturer	Dimension (in.)			Volume (in^3)	Weight (lb)
							Height	Width	Length		
CLBT 1	Gemini	45	24	Secondary	MAR4144-3	Eagle-Picher	6.4	7.9	5.6	283	20
CLBT 2	Gemini	440	24	Primary	MAP4138-3	Eagle-Picher	13.0	7.5	10.8	1050	117
CLBT 3	LEM	300	28	Secondary	MAR4273	Eagle-Picher	8.0	5.6	34.8	1580	125
CLBT 4	LEM	300	28	Primary	MAP4271	Eagle-Picher	8.0	5.6	37.2	1690	
CLBT 5	LEM	440	28	Primary	MAP4272	Eagle-Picher	10.0	8.2	17.5	1440	140

Note: Operating temperature of battery is +40° to +90 F

most of these performance characteristics and their upper and lower limits have been established as a result of prior tests and studies, the absolute lower limit of power system parasitic power demand has not been established. It has only recently been proven feasible that an Apollo Block II fuel cell can be operated, for at least several days without interruption, in a low-power mode with the positive terminal disconnected from the spacecraft d-c bus system. The aforementioned power system parasitic power demand includes fuel cell pumps, heaters, instrumentation, radiator heaters, CGSS (cryogenic gas supply system) heaters and blowers, solenoid valves in fluid transport lines.

To place a fuel cell in the powered-down or hot-standby condition, as it is sometimes called, the flight crew simply disconnects the fuel cell by momentarily placing the appropriate toggle switches in the OFF position. For instance, suppose that in an RCM laboratory equipped with three Apollo Block II fuel cells, the flight crew desires to place two fuel cells on hot standby. A crew member momentarily places the four associated Fuel Cell Main Bus ON/OFF toggle switches on the displays and controls panel in the OFF position. This will disconnect the positive output terminal of both fuel cells from the spacecraft main buses A and B. After this has occurred, the two fuel cells are on hot standby.

Apollo Block II Fuel Cell Performance Characteristics

Apollo Block II fuel cell projected performance estimates and Block I test data were analyzed to establish baseline Block II fuel cell performance characteristics. Except for minor updating changes, performance characteristics under normal operating conditions are the same as those reported in (Reference 14) unless otherwise stated. The estimated performance of the Block II fuel cell systems as reported in Reference 14 is summarized and reproduced in this section wherever advantageous.

Voltage-Power-Temperature Characteristics

Figure 77 is a composite graphical illustration of figures previously presented to NASA-MSC. The steady-state operating line was extended from 700 watts down to 563 watts gross power to cover requirements of the fuel cell procurement specification (Reference 15). Maximum gross power was increased from 1420 watts to 1500 watts, based on the Pratt-Whitney estimation [illegible]ted by Figure 15 of Reference 16. Constant-temperature voltage-power lines are identical to those of Figure 32 of Reference 16, with updated constant-temperature operating lines as reported in Reference 17. Characteristics of Figure 77 are utilized exclusively in this report to represent the Apollo Block II fuel cell. Fuel cell electrode-electrolyte temperature range for space or vacuum operation is estimated to be the same as those of

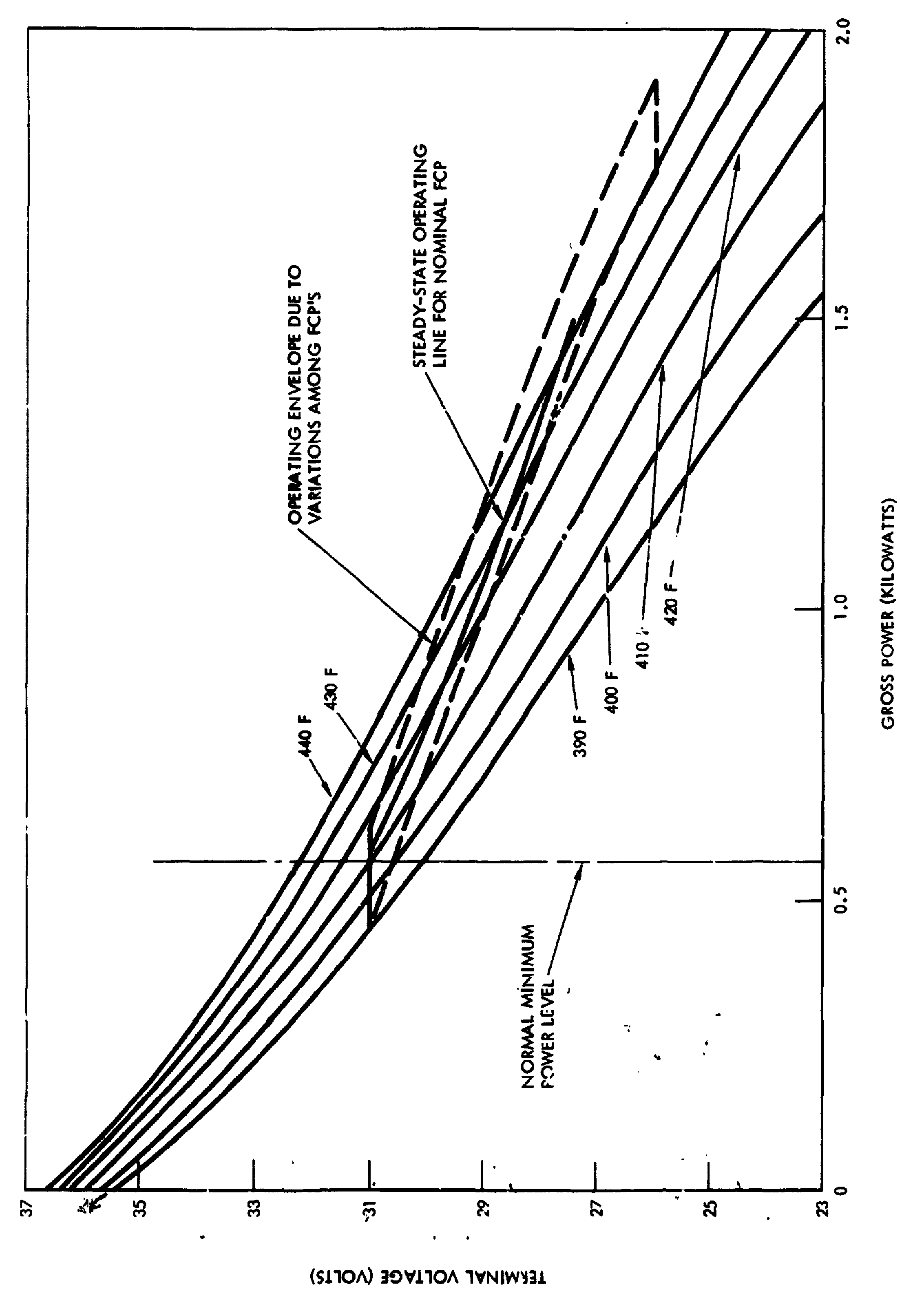


Figure 77. Power Characteristics of Apollo Block II Fuel-Cell Powerplant

Figure 77. Temperatures measured on the mid-cell in the stack, Cell No. 17, are estimated as approximately 20 F higher in space or vacuum operation than at sea level.

Load Sharing

The steady-state operating line of Figure 77 can be projected to a system steady-state operation of two and three fuel cell power plants (FCP) connected in parallel with the assumption that each FCP has identical performance characteristics. However this is not realistic since the performance characteristics of each FCP is expected to vary because of inherent design and manufacturing tolerances and because of variable operational conditions, especially temperature differences. The final result is unequal load sharing between FCP units of a system supplying a common spacecraft power requirement. Figure 77 indicates the normal range of FCP performance variation resulting from inherent design and manufacturing tolerances. This data is based on test measurements made by P&W in 1964 as part of the FCP development program and is shown superimposed on the constant-temperature operating lines. The difference in load sharing capability between two fuel cells selected at random is seen from the figure to be no greater than 100-150 watts, dependent to some extent upon voltage. Data from Figure 77 were utilized in estimating maximum power variation for two and three FCP systems. Table 43 shows the calculations at three power levels for a two FCP system when one FCP is at maximum performance and the other at minimum expected performance. Table 44 presents similar calculations for a three FCP system when the performances are one FCP at maximum, one FCP at minimum, and one FCP at average.

The results of the load sharing analysis are graphically presented in Figure 78. Variation in percent of load shared is seen by Figure 78 to be greatest at minimum power level, decreasing continuously as the power level of the combined fuel cell system is increased.

At minimum power level of 563 watts/FCP, maximum power variation is seen to be 150 watts. At maximum power level of 1500 watts/FCP, maximum variation is 130 watts. At intermediate levels between power limits, maximum variation is intermediate between 150 and 130 watts.

Transient Capabilities

The steady-state operating line shown in Figure 77 represents the focus of thermal equilibrium voltage versus power points for the FCP. Step changes in the mission power requirements will result in an operation along a constant temperature line from the initial power level to the new power level. At the new power level, the voltage will gradually approach the steady-state operational line as the FCP operating temperature approaches

Table 43. Maximum Power Variation Between Two Acceptable Fuel Cells

Fuel Cell No.	Power (kw)	Percent Total Power
1	0.6	57.2
2	0.45	42.8
Parallel combination of 1, 2	1.05	100.0
1	1.16	53.7
2	1.00	46.3
1, 2	2.16	100.0
1	2.13	51.6
2	2.00	48.4
1, 2	4.13	100.0

Table 44. Maximum Power Variation Among Three Acceptable Fuel Cells

Fuel Cell No.	Power (kw)	Percent Total Power
1	0.6	38.2
2	0.45	28.7
3	0.52	33.1
Parallel combination of 1, 2, 3	1.57	100.0
1	1.16	35.8
2	1.00	30.9
3	1.08	33.3
1, 2, 3	3.24	100.0
1	1.51	35.1
2	1.35	31.5
3	1.43	33.4
1, 2, 3	4.29	100.0
1	2.13	34.4
2	2.00	32.3
3	2.06	33.3
1, 2, 3	6.19	100.0

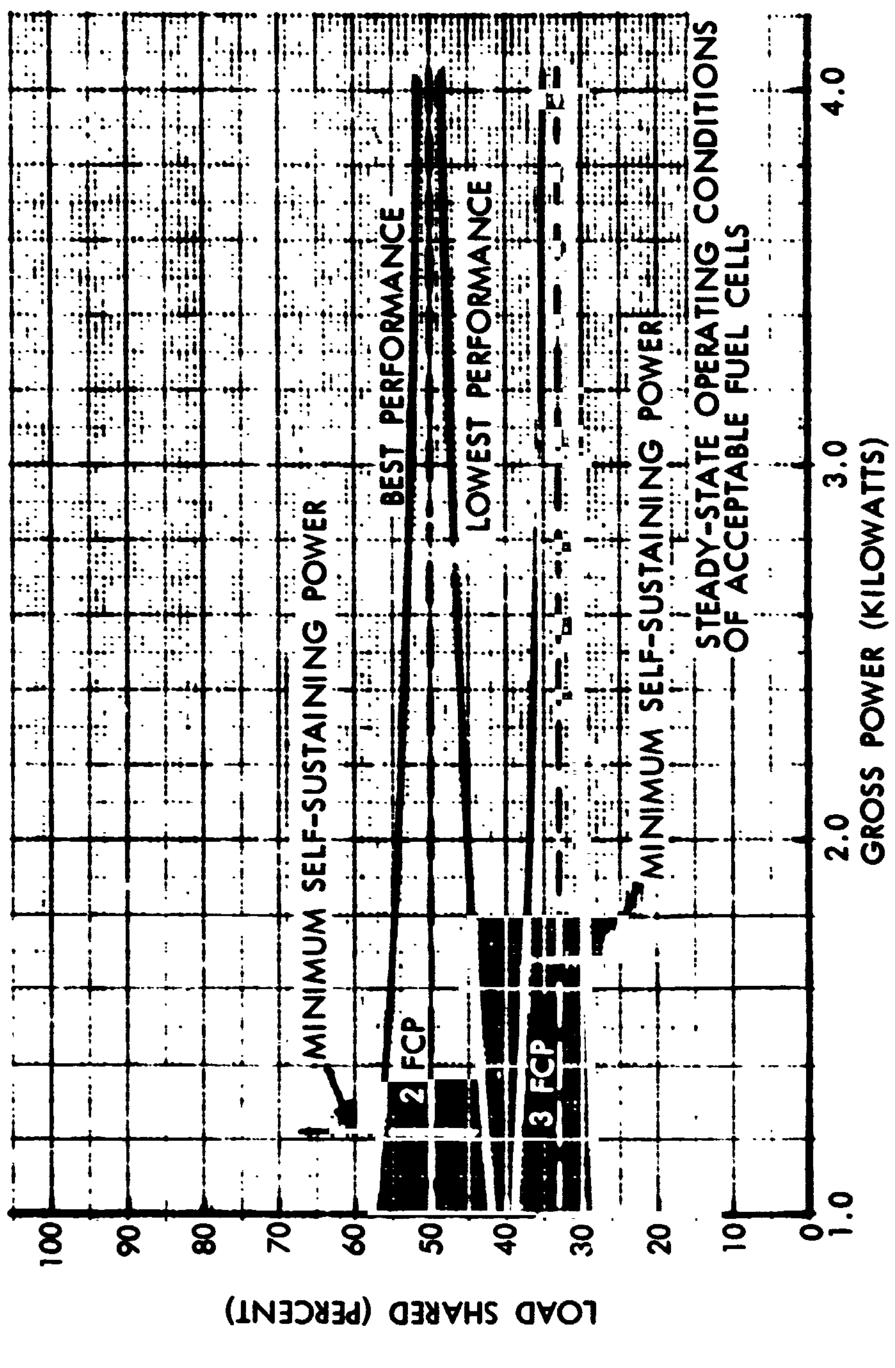


Figure 78. Load Sharing Between Fuel Cells

its equilibrium value. This transient delay is a function of the step power increment and the spacecraft thermal environment. Figure 79 shows a typical thermal lag of the FCP as the result of a step increase, followed by a step decrease in power. As indicated, up to three hours may be required to reach thermal equilibrium. This implies that seldom will the FCP be operating on the steady-state operating line. Instead, operation will normally be within the area enclosed by dashed lines shown in Figures 80 and 81. To maintain the system voltage within the voltage limitation of 27.0 to 31.0 volts, operation must be restricted from the shaded portions of the dashed line operational area. This can be accomplished by (1) restricting the range of step power increments and/or (2) using voltage limitation for step power decreases and peaking batteries for step power increases. Figure 82 defines the allowable transient power increment for a unit Block II FCP and this data can be projected to include two or three FCP operations. A ten-percent contingency factor was included in the calculations for defining transient capability.

Figures 80 and 81 were again based on the unrealistic assumption of identical average performance characteristics for the unit FCP's comprising the total system. These curves can be projected to include the variation between FCP units defined and shown in Figures 77 and 78.

Superimposed upon Figure 80, which graphically depicts transient and steady-state operation of a nominal or average performance of FCP systems, are individual variation data of Table 43 and Figures 77 and 78. This superposition results in the construction of Figure 83. Outer bounds of Figure 82 represent estimated maximum variations in voltage-power performance. Variations arise from two sources: (1) temperature induced transients as previously described; and (2) inherent variations among acceptable fuel cells. Suppose, for example, that the combination of two fuel cells is delivering 3.00 kilowatts of gross power. Further suppose that the fuel cell combination provides nominal, that is, average, identical performance characteristics. Voltage of the combination (Figure 83) is therefore 27.4 for this condition of steady-state operation. If the gross load shifts to 1.125 kilowatts (minimum self-sustaining limit), the voltage shifts along the dashed line of Figure 83 to 31.7 volts which is within the over-voltage region. If the load is then maintained constant at 1.125 kilowatts, the voltage gradually moves downward in the vertical direction at an exponential time rate as steady-state equilibrium is approached. The voltage finally stabilizes as 31.0 volts. This is at the opposite end or terminus of the nominal steady-state operating line of Figure 83. Vertical tie-lines are equally spaced in the figure. If the load is now increased back to the original load of 3.0 kilowatts, the voltage swings downward along the dashed line in the direction of the arrow to a point in the peaking battery region, below the minimum voltage of 27.0 volts as shown in the figure. The voltage then begins to rise according to the same exponential principle previously mentioned. Voltage rise at first is fairly rapid, then

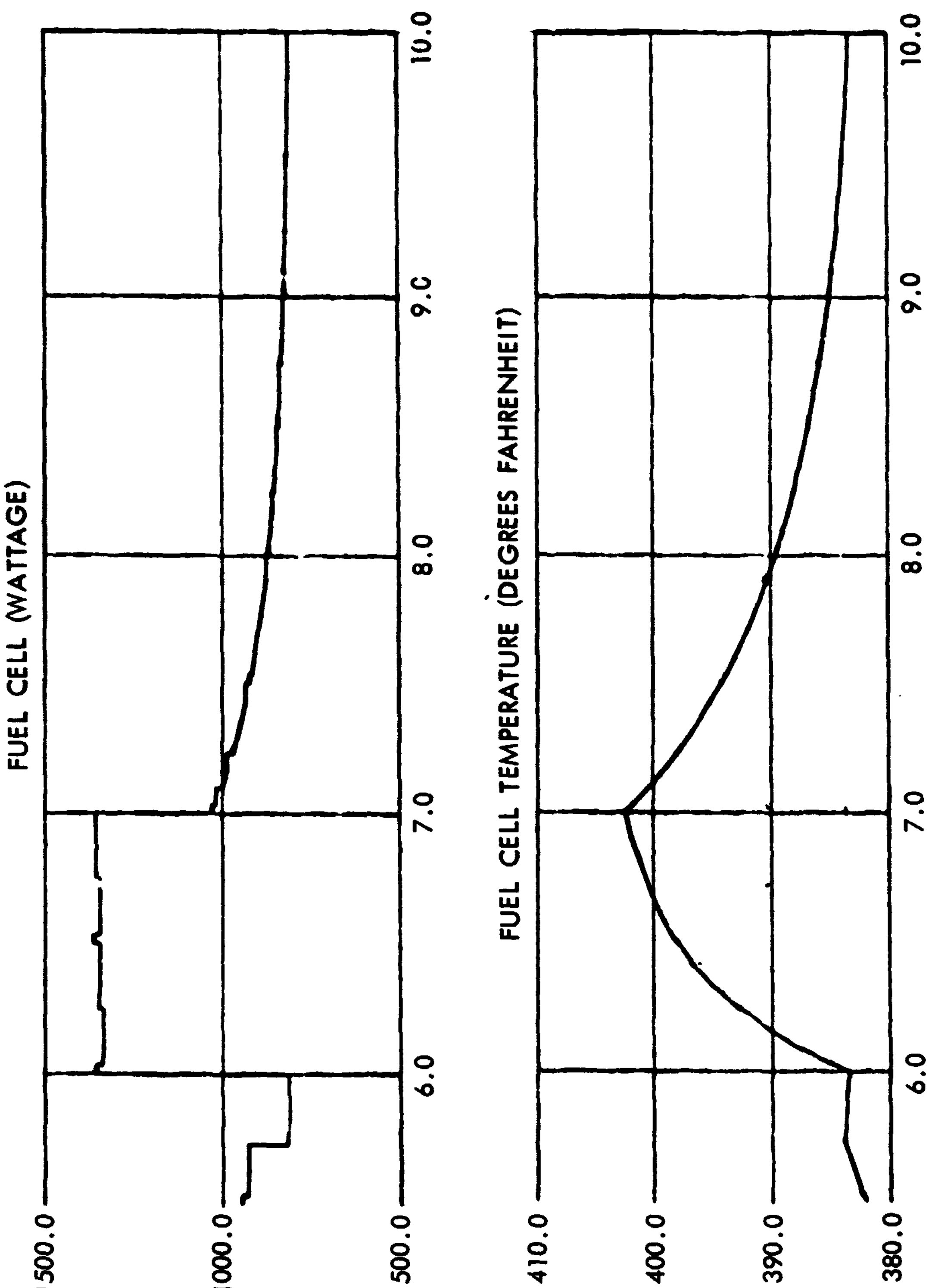


Figure 79. Temperature Characteristics of FCP Response to Step Power Change

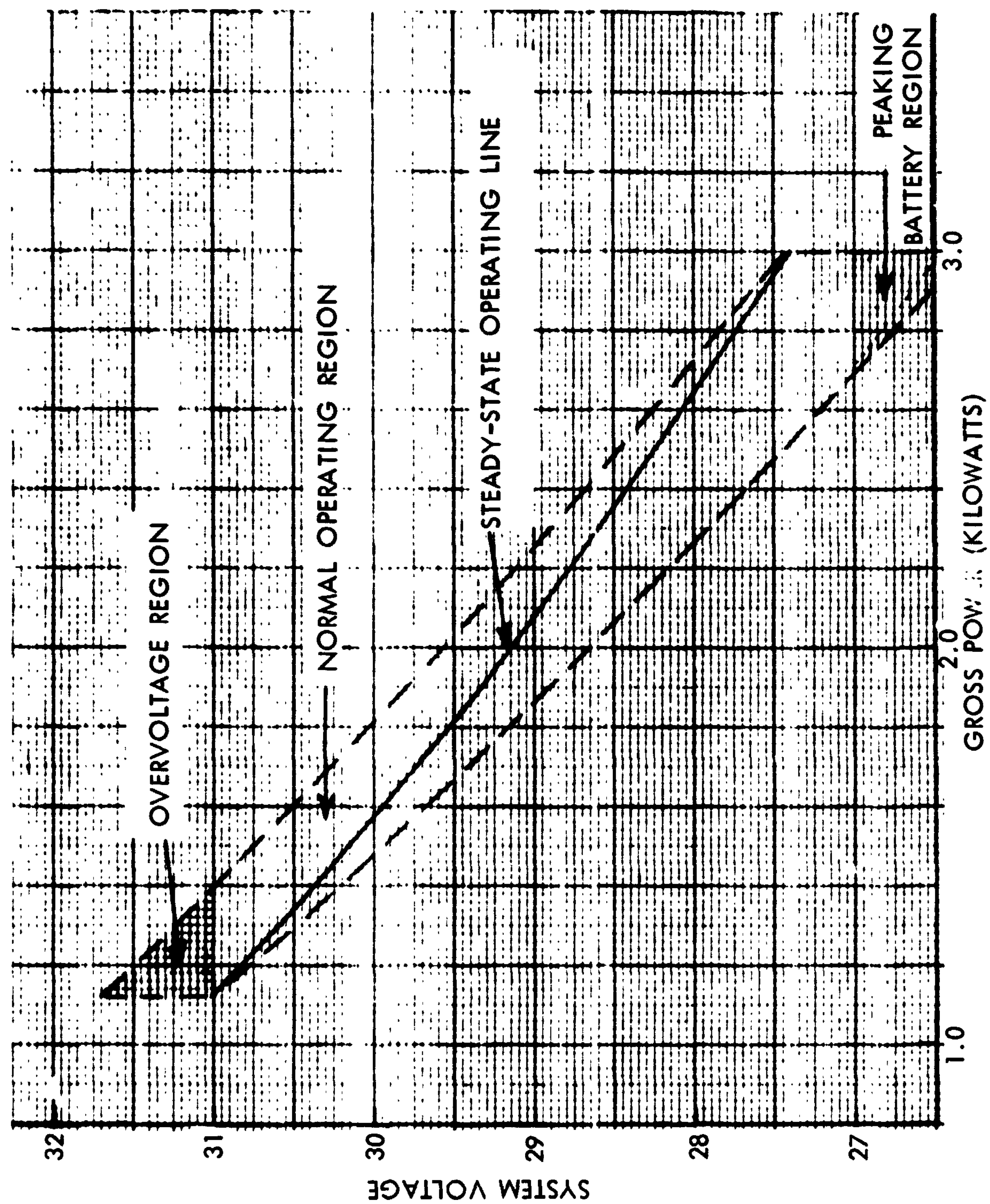


Figure 80. Operational Regions for Two FCP's in Parallel

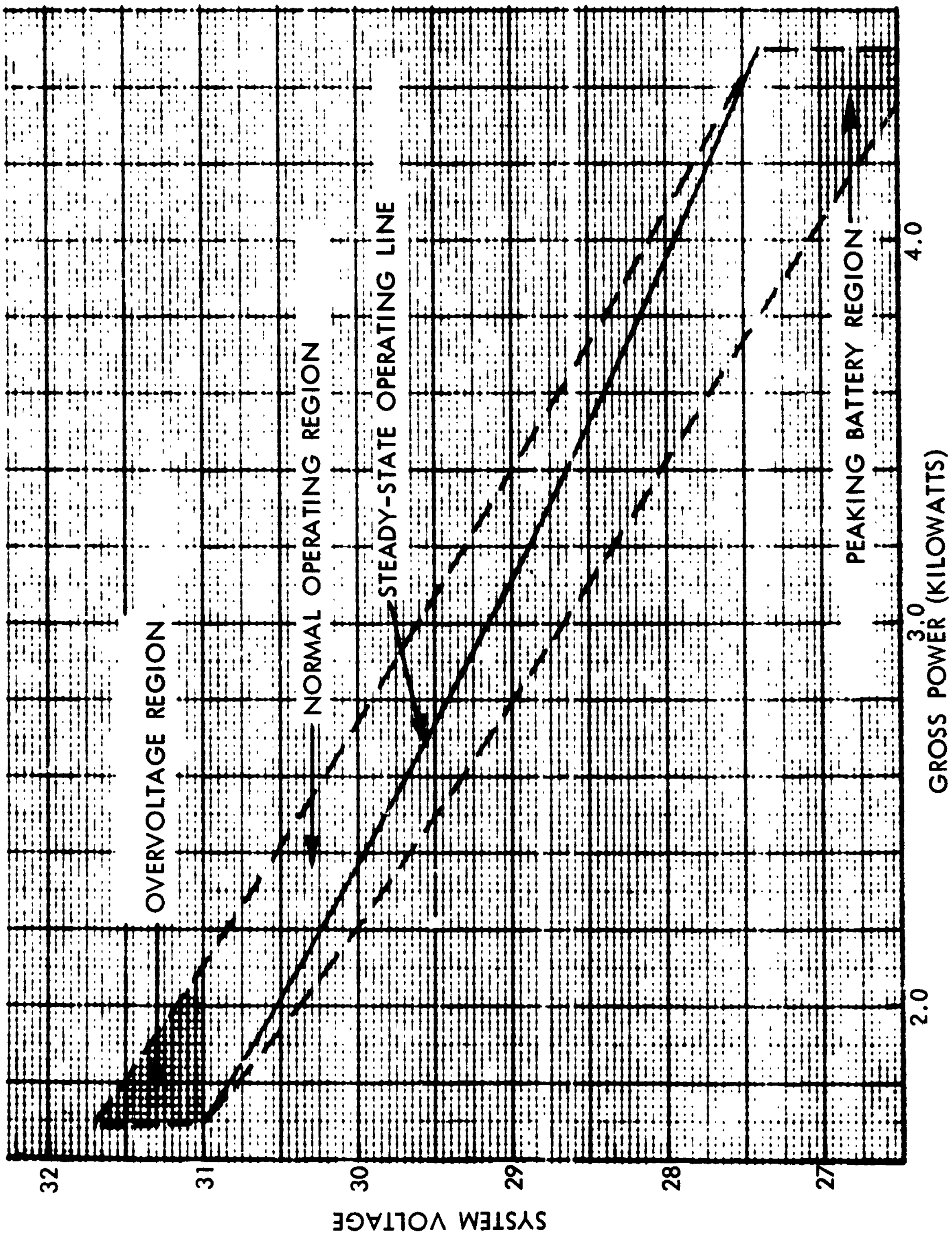


Figure 81. Operational Regions for Three FCP's in Parallel

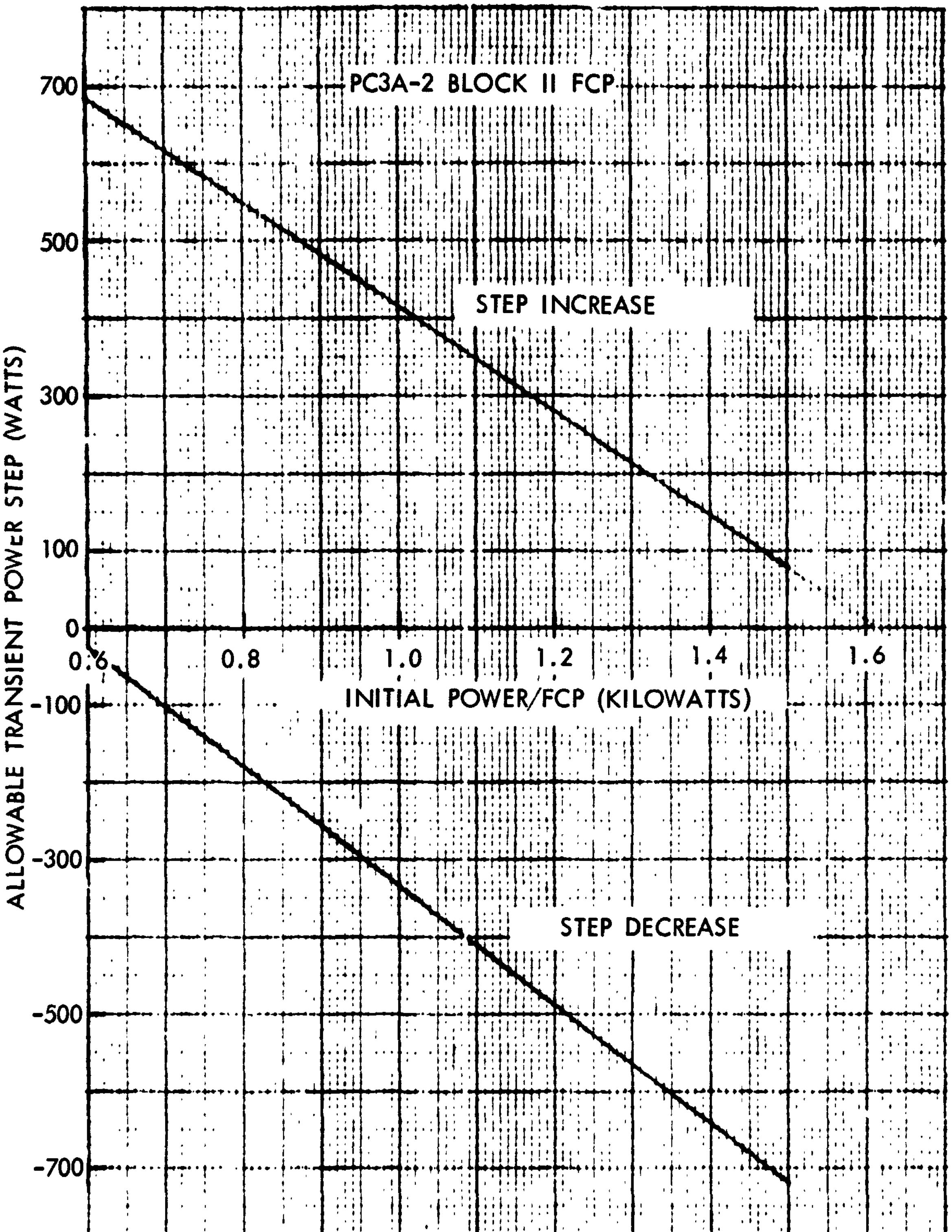


Figure 82. Estimated Transient Power Capability/FCP

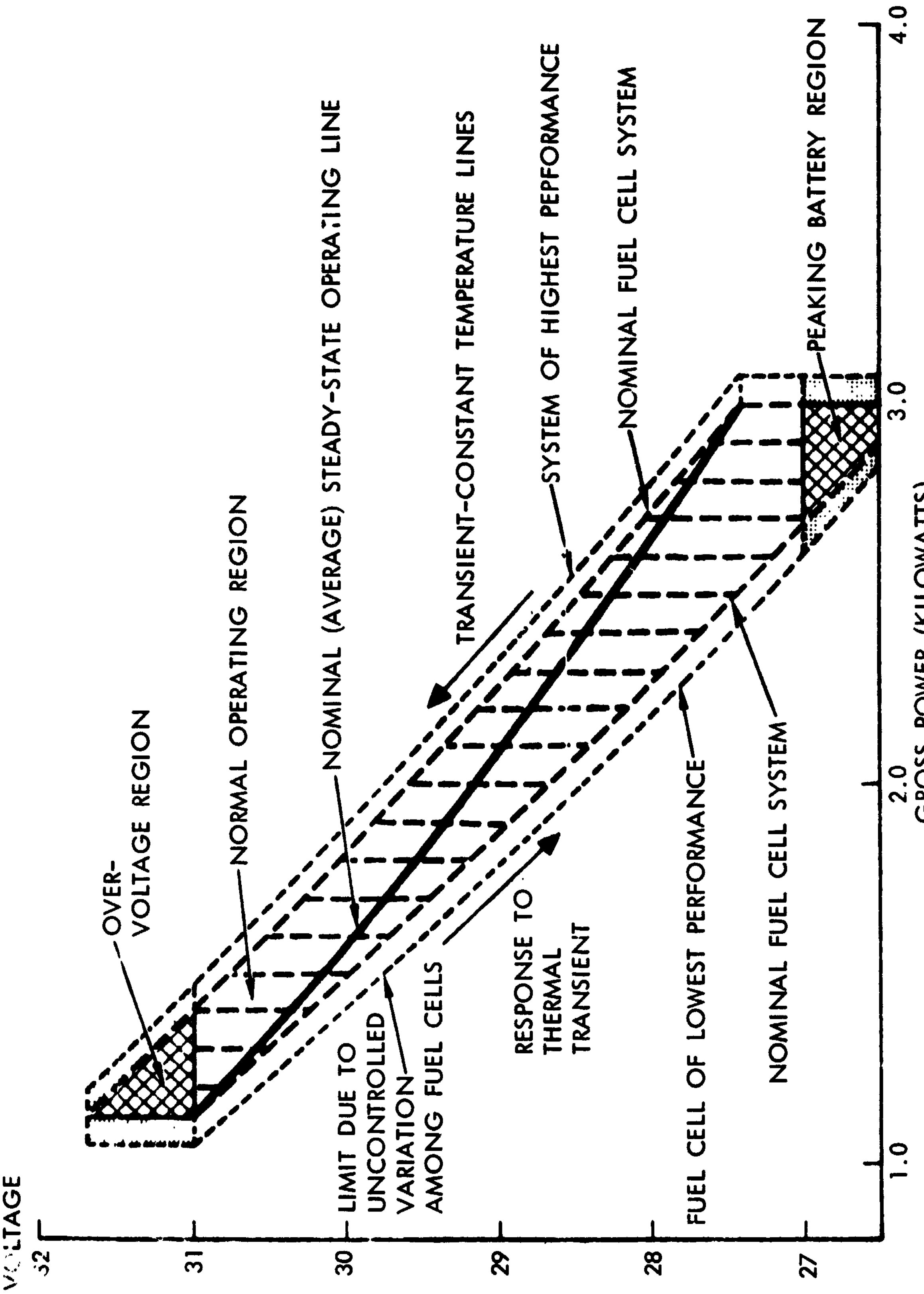


Figure 83. Thermal-Transient Performance, Two-FPC System

decreasing in rate as it approaches steady state. During this transient interval the voltage moves upward along the vertical tie-line, terminating finally at 27.4 volts, the point of origin.

The same principle as that described for the nominal performance system applies to the FCP of highest performance (dotted line with arrow in upward direction), or the FCP of lowest performance (dotted line with arrow in downward direction). Under no circumstances of load variation between minimum and maximum bounds of 1.045 and 3.075 kilowatts can voltage be outside the dotted boundary of Figure 83.

Figure 84 was constructed from data of Figures 77, 78, and 81 for a 3-FCP system in a manner completely analogous to that used for Figure 83. Just as was the case for the two FCP system, power and voltage are at all times constrained within the dotted line boundary. Also, voltage during a transient thermal lag interval moves along a vertical tie-line. Vertical tie-lines in Figure 84 are depicted for the nominal system only. To include the dotted line boundary, they may be extended as required.

Reactant Consumption

The reactant consumption characteristics for the baseline PC3A-2 Block II FCP were determined by utilizing the equations defined in Reference 16.

$$SPC\ (H_2) = \frac{0.0829}{V}\ \text{lb/kw-hr}$$

and

$$SPC\ (O_2) = 8\left[SPC\ (H_2)\right] = \frac{0.6632}{V}\ \text{lb/kw-hr}$$

where

V = volts/cell

Table 45 and Figure 85 give the results of the reactant consumption calculations on a unit FCP gross power basis.

Tables 46 and 47 and Figure 86 present the calculations for two and three FCP systems on a net power basis. The parasitic power load is estimated at 125 watts per FCP referenced to the 28 vdc SM bus. This estimate assumes that the new high-speed hydrogen pump-separator assembly will be utilized on the Block II FCP.

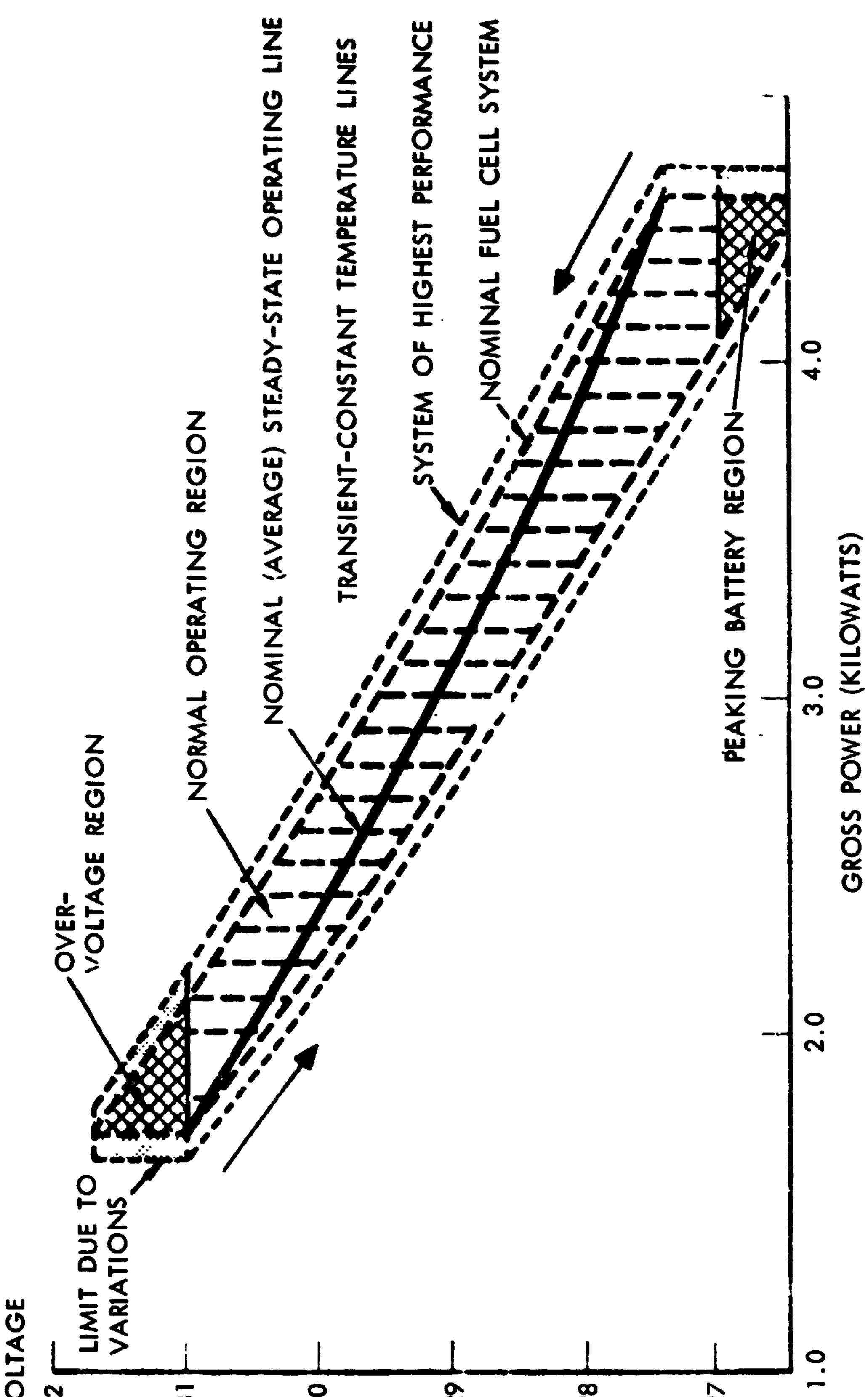


Figure 84. Thermal-Transient Performance, Three-FPC System

Table 45. FCP Reactant Consumption Calculation

Power (watts)	FCP (volts)	Volts/ Cell	SPC (H_2) (lb/kw-hr)	SPC (H_2) (lb/hr)	SPC (O_2) (lb/hr)	SPC ($H_2 + O_2$) (lb/hr)
563	31.00	1.00	0.0829	0.0466	0.373	0.420
800	29.95	0.966	0.0857	0.0686	0.548	0.617
1000	29.15	0.941	0.0881	0.0881	0.705	0.793
1200	28.42	0.918	0.0904	0.1084	0.869	0.977
1400	27.75	0.895	0.0925	0.1295	1.035	1.165
1500	27.40	0.884	0.0938	0.1405	1.125	1.266

Figure 86 indicates that, from a reactant consumption basis, two fuel cell operation is preferred for average net power levels below 2300 watts, while three fuel cell operation is preferable above 2300 average net watts. The difference between two and three FCP operation is not great when calculated on the basis of average power and operation on the steady-state operational line. The differences would become greater when transient operation is considered for any specific mission power-time profile, especially if the step power increments require operation in the volrage limiter or peaking battery regions as shown in Figures 83 and 84. Accurate determination of reactant consumption involves trade-off differences which would require computer simulation of the actual mission power profile. This program was developed during the PDP study under contract NAS 9-5017.

Water Production

The water production is assumed to be equal to the reactant consumption as shown in Figures 85 and 86.

Heat Rejection

The waste heat to be rejected by radiation to space through the EPS radiator subsystem was calculated as the difference between the higher heat value of the hydrogen gas consumed and the net power output of the FCP system. The electrical parasitic load per FCP is 125 watts referenced to the 28 vdc bus; however, not all of this must be rejected as waste heat by the EPS radiator. Approximately 30 watts (102 Btu/hr) represents the dc-ac inverter inefficiency, which is rejected through the ECS radiator system. The

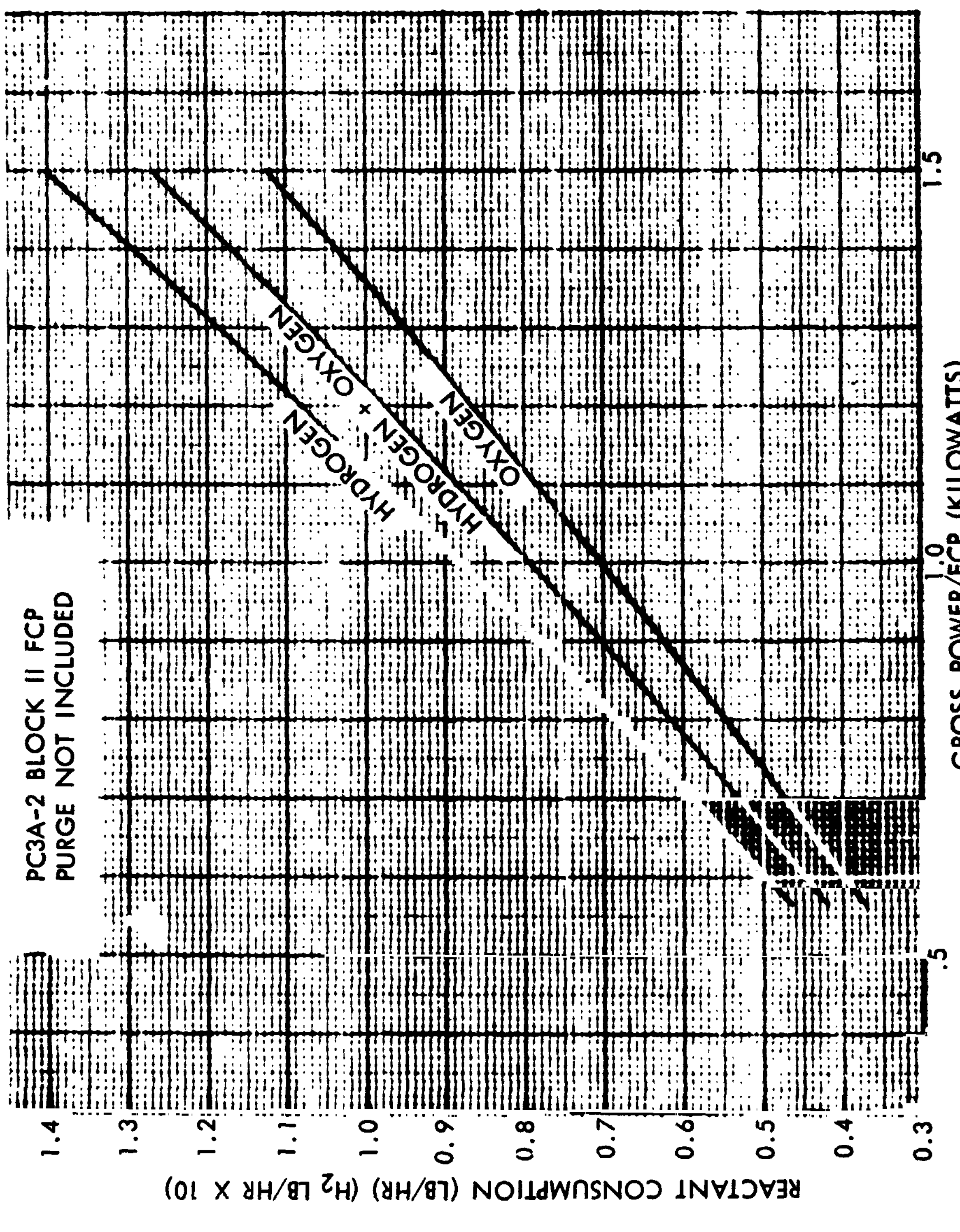


Figure 85. Estimated Reactant Consumption Per FCP

Table 46. Two-FCP System Reactant Consumption Calculations

Net Power (watts)	Gross Power (watts)	SPC (H_2) (lb/hr)	SPC (O_2) (lb/hr)	SPC ($H_2 + O_2$) (lb/hr)
876	1126	0.0932	0.747	0.840
1170	1420	0.122	0.976	1.098
1470	1720	0.148	1.184	1.332
1970	2220	0.196	1.568	1.764
2470	2720	0.250	2.000	2.250
2750	3000	0.281	2.248	2.529

Table 47. Three-FCP System Reactant Consumption Calculations

Net Power (watts)	Gross Power (watts)	SPC (H_2) (lb/hr)	SPC (O_2) (lb/hr)	SPC ($H_2 + O_2$) (lb/hr)
1315	1690	0.140	1.120	1.260
1455	1830	0.152	1.216	1.368
1955	2330	0.198	1.584	1.782
2455	2830	0.248	1.984	2.232
2955	3330	0.297	2.376	2.673
3455	3830	0.348	2.784	3.132
3955	4330	0.402	3.216	3.618
4125	4500	0.421	3.370	3.791

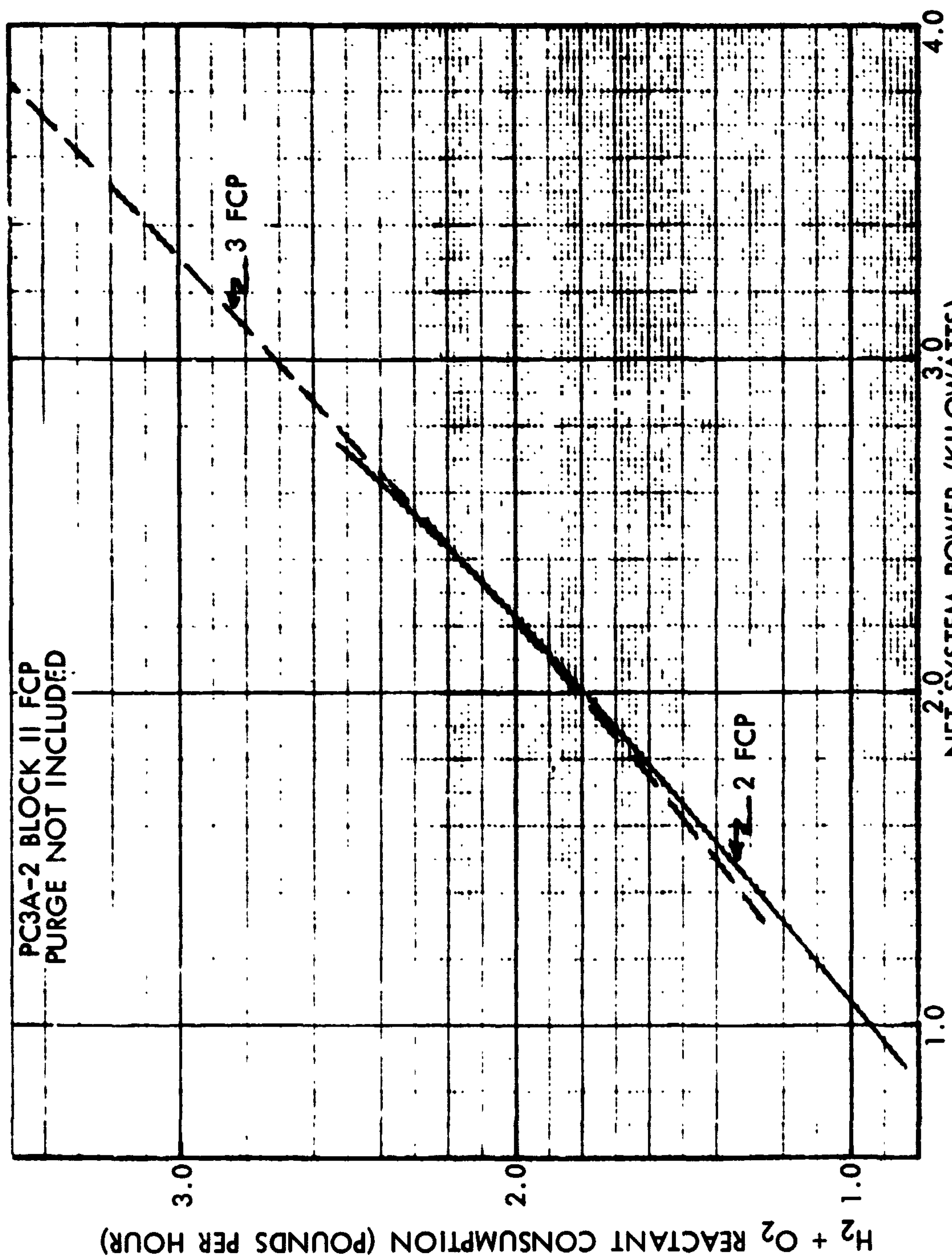


Figure 86. System Reactant Consumption ($H_2 + O_2$)

remaining 95 watts (325 Btu/hr) was assumed to be rejected by the EPS system. Figure 87 shows, on a bar graph, the basis used in calculating the heat rejection requirements. The FCP system efficiency was calculated as the ratio of net power to the heat equivalent of the consumed hydrogen gas, expressed in percent. Table 48 and Figure 88 present the heat rejection calculations on a unit FCP basis, while Table 49 and Figure 89 give the data for a two- and three-FCP system.

Voltage Drop and Power Loss in Transmission Lines

During transmission to inverter and spacecraft loads in the RCM Laboratory, distributed resistance in the transmission lines results in lowered voltage and loss of power. Using the same analytical techniques reported in Reference 14, Figure 90 was prepared depicting voltage drop as continuous functions of power in one, two, and three fuel cell systems.

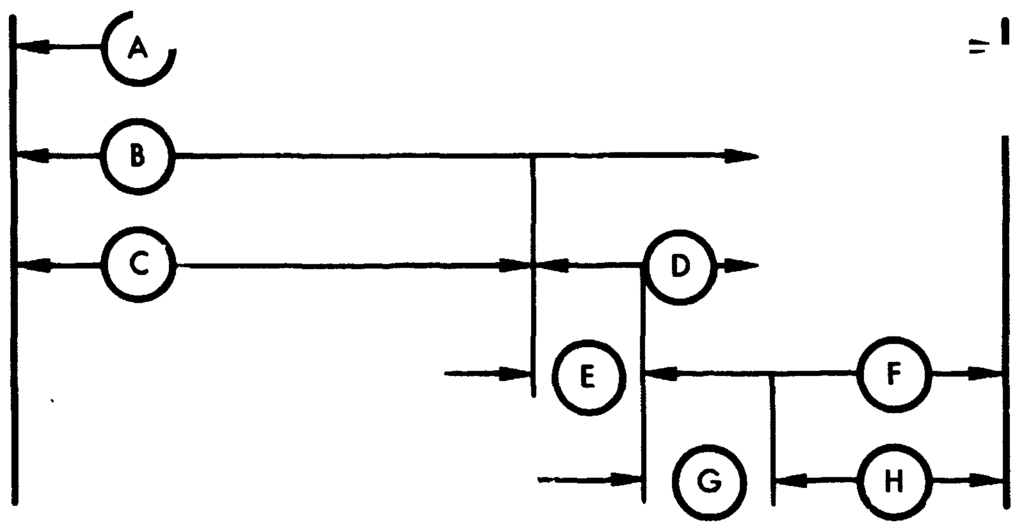
This figure was used to estimate voltage at the inverter input terminals, for conditions of high voltage, i.e., low power demand.

Hot Standby Mode of Operation

By disconnecting a fuel cell positive terminal from the two positive buses of the Apollo Block II power distribution system, the fuel cell is placed in the hot standby mode. During the ensuing cool-down period, the in-line heater temperature control sensor reaches 388 F, whereupon the in-line heater is automatically connected and the fuel cell begins to generate electric power. All electric power so generated is dissipated in the in-line heater, resulting in an increase of cell stack temperature. The temperature sensed by the in-line heater sensor also increases until, at 395 F, the in-line heater is disconnected.

According to results of the analytical heat-transfer study reported in Reference 18, the in-line heater may be expected to remain on 3.0 minutes, and stay off 1.0 minute to form a cycle period of 4.0 minutes. The study of Reference 18 was based upon many simplifying assumptions, including a local environment temperature of 80 F. Aforesaid numerical values of time periods are repeated here to provide data that are believed to represent operating conditions with reasonable accuracy. For example, the estimated 4.0-minute period might be determined by laboratory test to be as low as 2.0 or as high as 8.0 minutes; but test results cannot be reasonably expected to be outside of these limits.

Although tests of limited duration have been conducted at NASA/MSC and at Pratt-Whitney's Hartford, Connecticut facilities, never exceeding approximately two days of continual operation, test results have not been



where:

A = Hydrogen consumption heat equivalent

= H_2 (lb/hr) x 61,000 Btu

B = Gross electrical power output

C = Net electrical power to CSM loads

D = FCP parasitic power, 125 watts dc/FCP

E = DC-ac inverter inefficiency, heat rejected by ECS radiator subsystem

F = Heat rejected by EPS subsystem

A - (E + C). This is plotted in Figures XVI-C-13 and 14

G = Heat rejected to SM structure and environment

H = Heat rejected by EPS radiators

$$\text{Efficiency} = \frac{C}{A} \times 100$$

Figure 87. Heat-Rejection Calculation Definition

Table 48. Heat Rejection and Efficiency Per FCP

Net Power (watts)	Net Power (Btu/hr)	Gross Power (watts)	SPC (H_2) (lb/hr)	H_2 Heat Equivalent (Btu/hr)	Heat Rejection (Btu/hr)	Efficiency (Percent)
438	1494	563	0.0466	2840	1244	52.7
675	2300	800	0.0686	4185	1783	55.1
875	2984	1000	0.0881	5375	2289	55.6
1075	3670	1200	0.1084	6620	2848	55.6
1275	4350	1400	0.1295	7900	3448	55.1
1375	4690	1500	0.1405	8570	3788	54.7

published. It may be reasonably expected that such tests will be repeated and reported in the near future to support the AAP program.

Determination of Performance Characteristics, Limits and Constraints Relating to Bus-Disconnected Hot Standby Mode

Since each fuel cell while generating electric power requires a-c power for (1) water-separator pump, (2) radiator coolant (glycol-water) pump, (3) instrumentation, and requires d-c bus power for d-c instrumentation, at least one fuel cell or alternate d-c source must be connected to the spacecraft d-c bus system. Parasitic power system parameters which must be known are:

1. Power
2. Voltage
3. Inverter input voltage
4. Reactant consumption rate for fuel cells
5. Heat generation rate

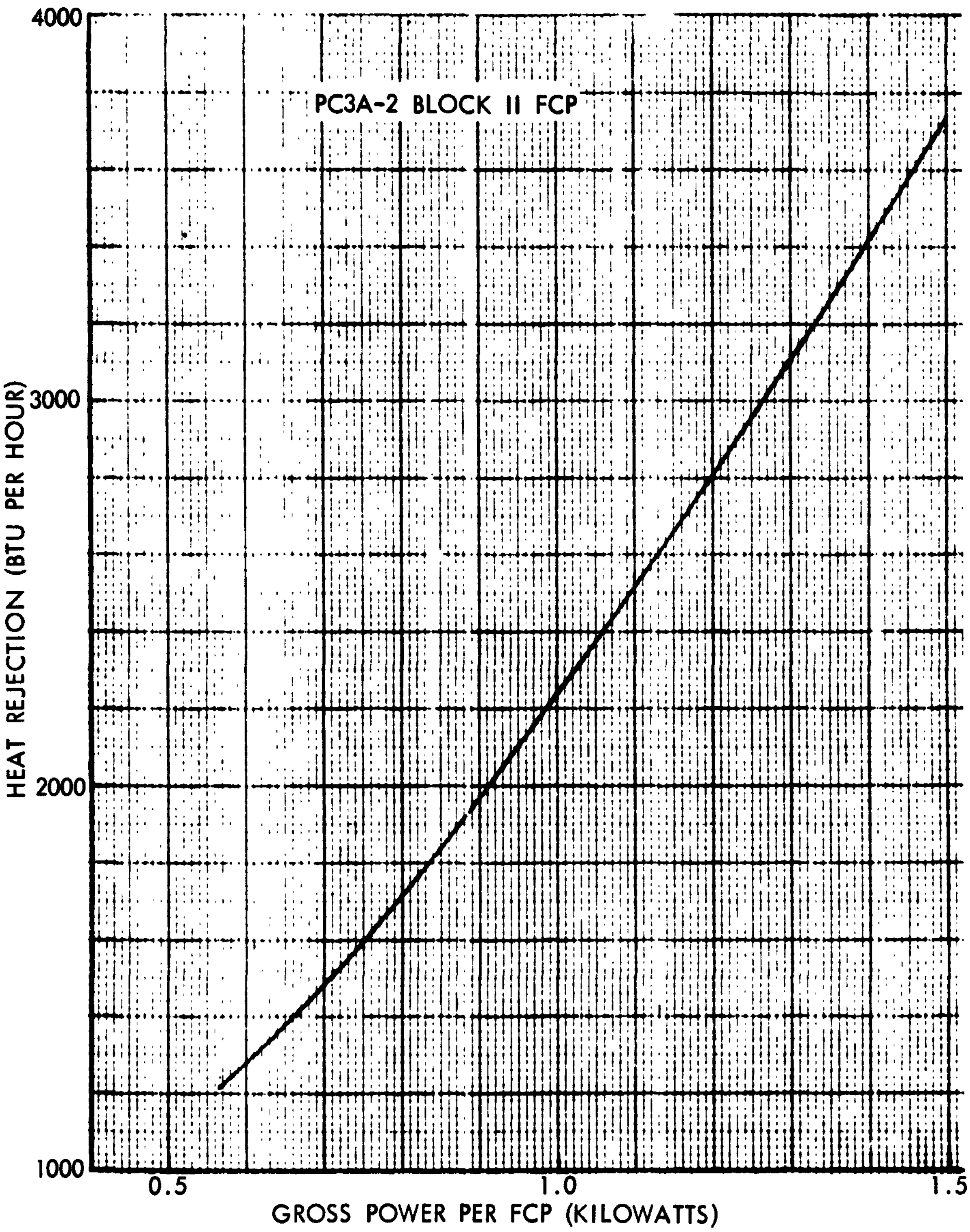


Figure 88. Estimated Heat Rejection Per FCP

Table 49. FCP System Heat Rejection and Efficiency

Net Power (watts)	Net Power (Btu/hr)	Gross Power (watts)	SPC (H_2) (lb/hr)	H_2 Heat Equivalent (Btu/hr)	Heat Rejection (Btu/hr)	Efficiency (Percent)
TWO PC3A-2 BLOCK II FCP IN PARALLEL						
876	2988	1126	0.0932	5680	2488	52.7
1170	3992	1420	0.122	7440	3244	53.7
1470	5018	1720	0.148	9030	3808	55.6
1970	6723	2220	0.196	11950	5023	56.2
2470	8428	2720	0.250	15150	6518	55.6
2750	9378	3000	0.281	17120	7538	54.7
THREE PC3A-2 BLOCK II FCP IN PARALLEL						
1315	4482	1690	0.140	8540	3752	52.6
1455	4967	1830	0.152	9270	3997	53.6
1955	6672	2330	0.198	12080	5102	55.2
2455	8377	2830	0.248	15120	6437	55.4
2955	10067	3330	0.297	18110	7737	55.5
3455	11777	3830	0.348	21220	9137	55.4
3955	13477	4330	0.402	24550	10767	54.9
4125	14057	4500	0.421	25650	11287	54.7

All of the foregoing parameters have been established for conditions before launch, during ascent, and during space flight, for fuel cell systems comprising one, two, and three fuel cell systems, using the same or similar analytical techniques which have proven to be successful in prior studies relating to extended Apollo applications.

Results of the analytical calculations are summarized in Table 50 through 57. These tables show a systematic determination of load demand, transmission power loss, voltage, power, reactant consumption and heat generation for conditions of maximum, minimum, and average power generation.

During startup, excess heat generated by the fuel cells is passed on by the radiator to the inner region of the SLA, where it can be removed by the GSE forced-air cooling system which is available. Heat which must be removed during the prelaunch time interval is listed in Table 55, for one, two, and three fuel cell systems.

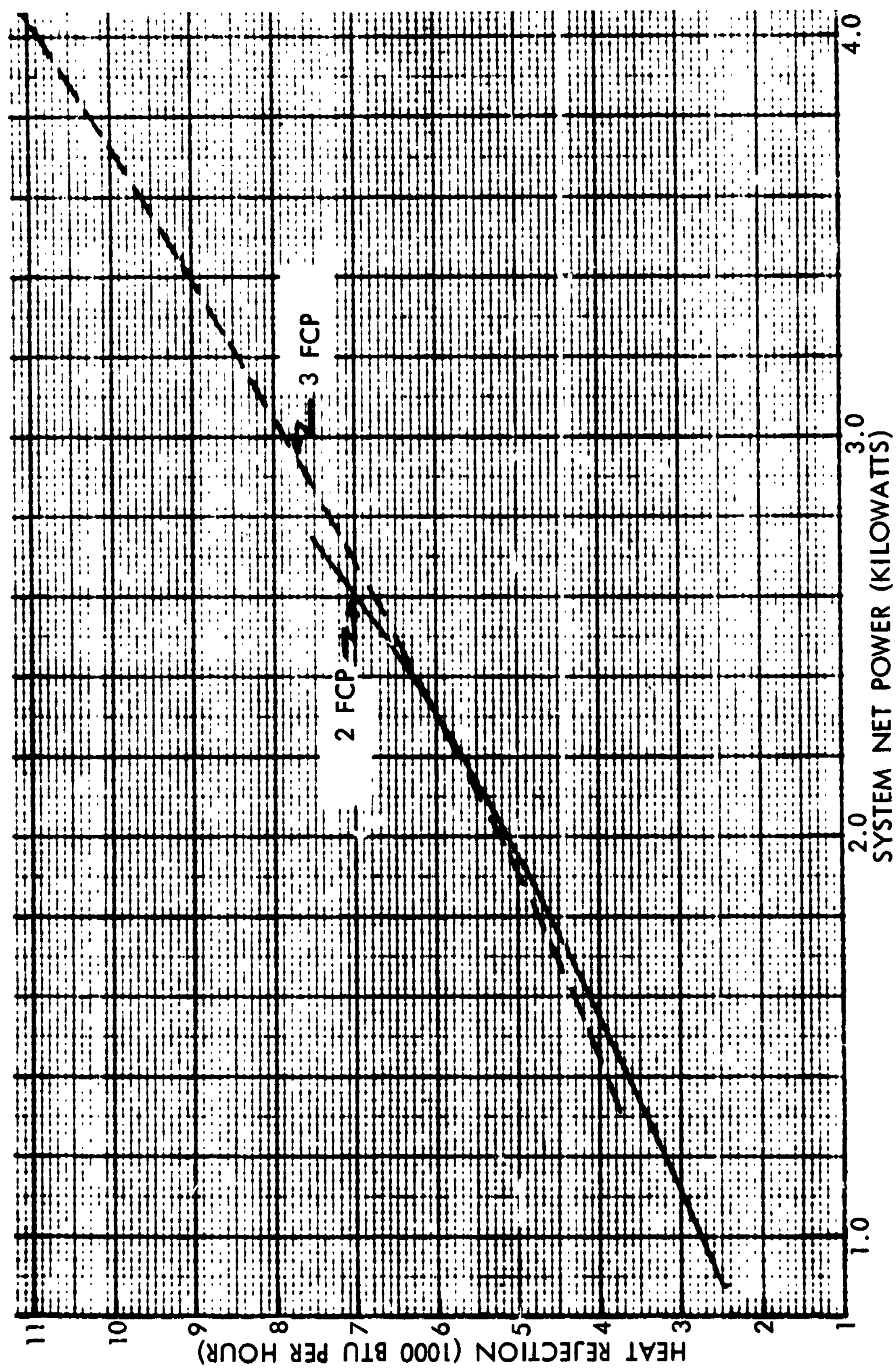


Figure 89. Estimated System Heat Rejection

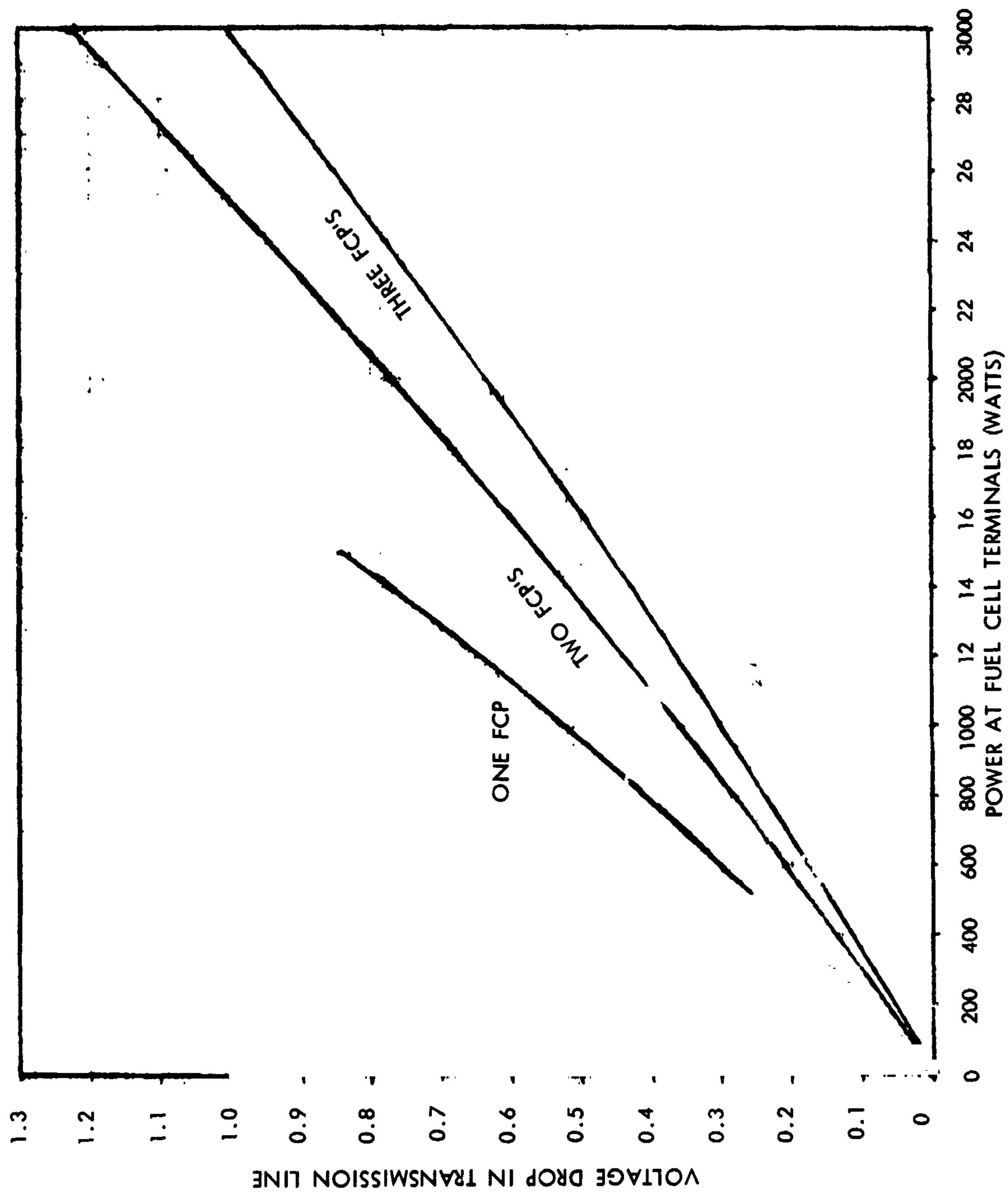


Figure 90. Line Voltage Drop During One-, Two-, and Three-Fuel Cell

Table 50. Power and Voltage Conditions of Bus-Connected Fuel Cell in Two-Fuel-Cell System During Prelaunch Time Period on Pad

Item	D-C POWER CONDITIONS (WATTS)				A-C POWER CONDITIONS (WATTS)			
	Connected	Duty Cycle (%)	Average	Continuous	Connected	Duty Cycle (%)	Average	Continuous
In-line heater	210.0	75.0	157.5					
Fuel cell instruments	17.0	100.0	17.0	17.0	1.3	100.0	1.3	1.3
Coolant pumps (1/FCP)					60.0	100.0	60.0	60.0
Water separator pumps (1/FCP)					90.0	100.0	90.0	90.0
Total a-c load					151.3	100.0	151.3	151.3
Inverter, 75% efficiency	202.0	100.0	202.0	202.0				
Total d-c load	429.0	88.0	376.5	219.0				
	D-C VOLTAGE CONDITIONS (VOLTS)							
Voltage at FCP terminals	31.1		31.6	33.2				
Voltage at inverter input terminals	30.9		31.4	33.1				

Note: During prelaunch time period, CGSS heaters and blowers are powered by GSE.

Table 51. Parasitic Loads of Bus-Connected Fuel Cell and Cryogenic Gas System (CGSS) in Two-Fuel-Cell System During Prolonged Time Period Wherein No Useful Power Is Delivered

Item	D-C Loads (watts)				A-C Loads (watts)			
	Connected*	Duty Cycle (%)	Average	Continuous	Connected*	Duty Cycle (%)	Average	Continuous
In-line heater	185.0	75.0	139.0					
H_2 tank heaters (1/Tank)	(2)60.0 (4)120.0	11.7	7.0 14.0					
O_2 tank heaters (1/Tank)	(2)370.0 (4)740.0	10.3	38.0 76.0					
Purge valves (2/FCP)	34.0		0.1					
Fuel cell instruments	16.7	100.0	16.7	16.7	1.3	100.0	1.3	1.3
Coolant pumps (1/FCP)					60.0	100.0	60.0	60.0
Water separator pumps (1/FCP)					90.0	100.0	90.0	90.0
H_2 tank blowers (1/Tank)					(2)100.0 (4)200.0	11.7 11.7	11.7 23.4	
O_2 tank blowers (1/Tank)					(2)280.0 (4)560.0	10.3 10.3	28.8 57.6	
Total a-c load					(2)531.3 (4)911.3	36.0 25.5	191.8 232.3	151.3 151.3
Inverter, 75% efficiency	(2) 710.0 (4)1215.0	36.0 25.5	256.0 310.0	202.0 202.0				
Total d-c load	(2)1375.7 (4)2310.7	33.2 24.0	456.9 555.8	218.7 218.7				

Note: Time of in-line heater duty cycle is estimated as 3.0 minutes on, 1.0 minute off.
*Number in parentheses indicates the number of pairs of cryogenic tanks (H_2 and O_2).

Table 52. Power and Voltage Conditions of Bus-Connected Fuel Cell in Three-Fuel-Cell System During Prelaunch Time Period on Pad

Item	D-C POWER CONDITIONS (WATTS)				A-C POWER CONDITIONS (WATTS)			
	Connected	Duty Cycle (%)	Average	Continuous	Connected	Duty Cycle (%)	Average	Continuous
In-line heater	190.0	75.0	142.0					
Fuel cell instruments	25.0	100.0	25.0	25.0	2.0	100.0	2.0	2.0
Coolant pumps (1/FCP)					90.0	100.0	90.0	90.0
Water separator pumps (1/FCP)					135.0	100.0	135.0	135.0
Total a-c load					227.0	100.0	227.0	227.0
Inverter, 75% efficiency	303.0	100.0	303.0	303.0				
Total d-c load	518.0	91.0	470.0	328.0				
	D-C VOLTAGE CONDITIONS (VOLTS)							
Voltage at FCP terminals	30.4	91.0	30.75	32.0				
Voltage at inverter input terminals	30.1	91.0	30.5	31.8				

Note: Time period of in-line heater duty cycle is estimated as 3.0 minutes ON, 1.0 minute OFF.

Table 53. Parasitic Loads of Bus-Connected Fuel Cell and Cryogenic Gas System (CGSS) in Three-Fuel-Cell System During Prolonged Time Period Wherein No Useful Power Is Delivered

Item	D-C Loads (watts)				A-C Loads (watts)			
	Connected	Duty Cycle (%)	Average	Continuous	Connected	Duty Cycle (%)	Average	Continuous
In-line heater	180.0	75.0	135.0					
H_2 tank heaters (1/Tank)	(2) 60.0 (4)120.0	11.7	7.0 14.0					
O_2 tank heaters (1/Tank)	(2)370.0 (4)740.0	10.3	38.0 76.0					
Purge valves (2/FCP)	34.0		0.1					
Fuel cell instruments	25.0	100.0	25.0	25.0	2.0	100.0	2.0	2.0
Coolant pumps (1/FCP)					90.0	100.0	90.0	90.0
Water separator pumps (1/FCP)					135.0	100.0	135.0	135.0
H_2 tank blowers (1/Tank)					(2)100.0 (4)200.0	11.7	11.7 23.4	
O_2 tank blowers (1/Tank)					(2)280.0 (4)560.0	10.3	28.8 57.6	
Total a-c load					(2)607.0 (4)987.0	44.0 31.2	267.5 308.0	227.0 227.0
Inverter, 75% efficiency	(2) 810.0 (4)1315.0	44.0 31.2	356.0 410.0	303.0 303.0				
Total d-c load	(2)1479.0 (4)2414.0	38.0 27.3	561.0 660.1	328.0 328.0				

Note: Time period of in-line heater duty cycle is estimated as 3.0 minutes ON, 1.0 minute OFF.
*Number in parentheses indicates the number of pairs of cryogenic tank (H_2 and O_2).

Table 54. Power and Heat Generating Parameters of Bus-Connected Fuel Cell During Prelaunch Period on Pad

Fuel cells per system	One	Two	Three
Fuel cells connected to d-c bus	One	One	One
D-c bus power (ave. watts)	275.5	376.5	470.0
Bus voltage (ave. volts)	32.5	31.6	30.7
Bus voltage per cell (volts)	1.050	1.020	0.990
Specific hydrogen consumption (lb/kw-hr)	0.0789	0.0811	0.0836
Hydrogen consumption rate (ave. lb/hr)	0.0217	0.0306	0.0393
Heat generation rate (ave. Btu/hr)	1,320.	1,865.	2,400.

Table 55. Power and Heat Generating Parameters of Bus-Disconnected Fuel Cells During Prelaunch Period on Pad or After Launch

Fuel cells per system	Two	Three
Fuel cells disconnected from d-c bus	One	Two
Connected (watts)	220.0	440.0
Duty cycle (%)	75.0	75.0
Average (watts)	165.0	330.0
Fuel cell voltage while generating power (volts)	33.1	33.1
Voltage/cell while generating power (volts)	1.068	1.068
Specific hydrogen consumption (lb/kw-hr)	0.0776	0.0776
H_2 consumption rate (ave. lb/hr)	0.0128	0.0256
Rate of heat generation (Btu/hr)	780.0	1,560.0

Table 56. Heat Generation and Reactant Consumption by Fuel Cell System During Prelaunch Time Period on Pad

Fuel cells per system	One	Two	Three
Fuel cells connected to bus	One	One	One
Fuel cells disconnected from bus		One	Two
Average rate of heat generation by bus-connected fuel cells (Btu/hr)	1,320	1,865	2,400
Average rate of heat generation by bus-disconnected fuel cells (Btu/hr)		780	1,560
Average rate of heat generation by fuel cell system (Btu/hr)	1,320	2,645	3,960
Rate of hydrogen consumption (ave. lb/hr)	0.217	0.0434	0.0649
Rate of oxygen consumption (ave. lb/hr)	0.1736	0.3472	0.5192

If it is desired to extend flight time by reducing reactant consumption, hence electric power generation, the hot standby mode can be used. It provides maximum extension of time since it requires less reactant consumption than any other operating mode.

While the fuel cell system is in hot standby mode, the bus-connected fuel cell can provide voltage at inverter input terminals which exceeds the specified allowable voltage of 30.0 if no other load is on the bus. Inverter input voltage was determined for maximum, minimum, and average power conditions of one, two, and three fuel cell systems operating in hot standby before launch and during space flight, for (1) a two-pair Block II cryogenic gas storage system (CGSS), and (2) a four-pair Block II CGSS. Highest voltage possible for any configuration and set of conditions is estimated to be 34.3. This is obtained from Table 57 as follows. If no useful power is drawn from a single fuel cell system during the prelaunch period, the dc bus voltage is 34.5. Allowing a 0.2-volt loss in transmission, the inverter input voltage is then estimated as 34.3 volts. For the same set of operating conditions, a single fuel cell system provides average voltage of 32.3 as estimated from average bus voltage of 32.5 (Table 54). These upper limits of inverter input voltage do not appear likely to damage the inverter. This can definitely be determined either from the manufacturer, Westinghouse Electric Corporation, or by test qualification of the entire hot standby operating procedure, including inverter operation.

Table 57. Power Parameters (Power, Voltage, Reactant Consumption, Heat Rejection) of One, Two, and Three Fuel Cell Systems with Two- and Four-Pair Cryogenic Tank Systems During Prolonged Time Period Wherein No Useful Power Is Delivered

Fuel Cells Per System		One			Two			Three		
Power Condition		Maximum			Minimum			Average		
Load power (watts)	(2)*	1267	109	348	1596	219	622	1919	328	891
	(4)	2202	109	447	2531	219	721	2854	328	990
D-c bus power (watts)	(2)	1267	109	348	1376	219	457	1479	328	561
	(4)	2202	109	447	2311	219	556	2414	328	660
Power loss in transmission (%)	(2)	25	1	4	32	2	6	36	4	12
	(4)	73	1	5	87	2	7	95	4	13
Power at bus-connected FCP terminals (watts)	(2)	1292	110	352	1408	221	463	1515	332	573
	(4)	2275	110	452	2398	221	563	2509	332	673
D-c bus voltage	(2)	27.7	34.5	31.8	27.3	33.1	30.8	27.0	32.0	30.0
	(4)	23.0	34.5	30.9	22.0	33.1	30.0	21.0	32.0	29.7
Bus voltage/cell	(2)	0.895	1.110	1.025	0.880	1.066	0.994	0.870	1.031	0.967
	(4)	0.741	1.110	0.996	0.710	1.066	0.969	0.678	1.031	0.959
Specific H_2 consumption by bus-connected FCP (lb/kw-hr)	(2)	0.0925	0.0745	0.0807	0.0941	0.0777	0.0834	0.0952	0.0801	0.0856
	(4)	0.1116	0.0745	0.0831	0.1167	0.0777	0.0855	0.1222	0.0801	0.0865
Specific H_2 consumption by bus-disconnected FCP (lb/kw-hr)	(2)				0.1036	0.0000	0.0766	0.1036	0.0000	0.0776
	(4)				0.1036	0.0000	0.0766	0.1036	0.0000	0.0776
Rate of H_2 consumption by fuel cell system (lb/hr)	(2)	0.1170	0.0081	0.0281	0.1730	0.0170	0.0650	0.2280	0.0263	0.1021
	(4)	0.2450	0.0081	0.0372	0.3180	0.0170	0.0743	0.3935	0.0263	0.1114
EPS heat rejection rate for bus-connectd FCP (But/hr)	(2)	3000	240	770	3430	490	[illegible]	3740	740	1220
	(4)	5900	240	1000	5900	490	[illegible]	6200	740	1420
EPS heat rejection rate** for bus-disconnected FCP (Btu/hr)	(2)				1040	0	780	2080	0	1560
	(4)				1040	0	780	2080	0	1560
EPS heat rejection rate** for fuel cell system (Btu/hr)	(2)	3000	240	770	4470	490	1800	5820	740	2780
	(4)	5900	240	1000	6940	490	1990	8280	740	2980

*Number in perentheses indicates the number of pairs of cryogenic tanks. One pair comprises one hydrogen tank and one oxygen tank.
**Comprises controlled heat rejection through radiator and uncontrolled heat rejection through insulation to local environment.

Possible freezing of water-glycol coolant in the radiator under extreme conditions of space flight has been examined. Minimum heat rejection rates have been computed, with results as listed in Table 57. With the recent addition of 5.0-watt electric heaters to each radiator coolant loop, coolant freezing is not expected to occur under any set of operating conditions which have been considered.

Water accumulation is expected to proceed without difficulty. All tests to date have demonstrated the practical feasibility of the water collection system. Upon examination, no reasons were found to support any hypothesis that hot standby would adversely affect the accumulation of water.

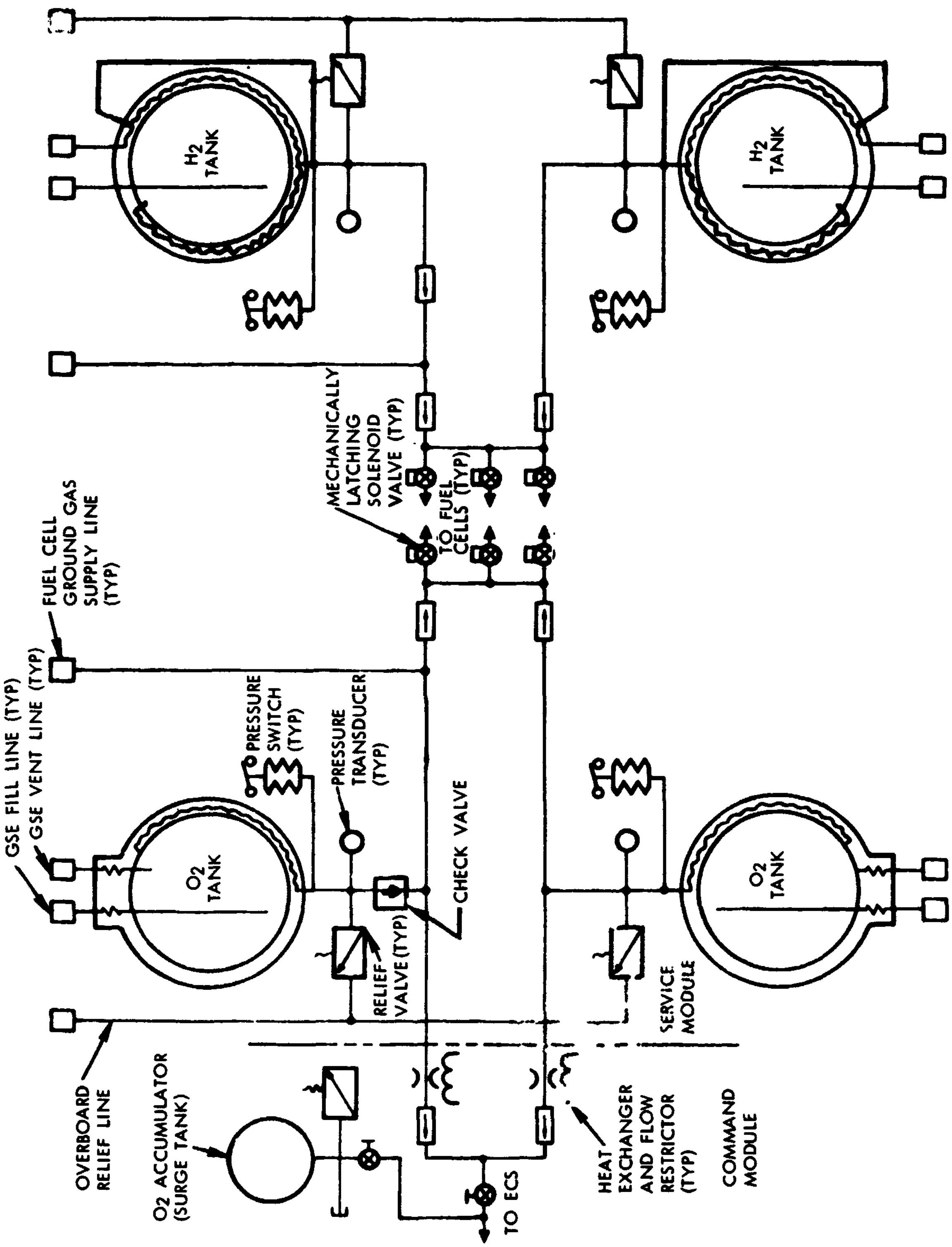
Apollo Block II Cryogenic Gas Storage System (CGSS) Components

Apollo Block II CGSS tanks are available in single units, as pairs comprising one oxygen tank and one hydrogen tank per pair, and as multiples of pairs. In any combination, the RCM Laboratory CGSS will comprise one or more hydrogen tanks, one or more oxygen tanks, hydrogen and oxygen valve modules, servicing disconnects, density and temperature signal conditioners, electrical connectors, fuel cell valve module for each fuel cell, and related plumbing. Convenient modules can be formed from multiples of tank pairs, as for example, a two-pair system comprising two hydrogen tanks and two oxygen tanks.

Block II Cryogenic Gas Storage System

The Block II cryogenic gas storage system includes two equally sized spherical oxygen tanks and two equally sized spherical hydrogen tanks. The storage tanks and associated controls are located in Sector IV of the service module. The tanks and plumbing are arranged so that oxygen can be supplied to the environmental control subsystem (ECS) from each oxygen tank. Oxygen and hydrogen can be supplied to any of the three fuel cells from each tank as shown schematically in Figure 91. The oxygen tanks and the hydrogen tanks and their controls are packaged into two complete and separate modules to permit final assembly and checkout of each system prior to final installation of the system in the spacecraft. The oxygen system module and the hydrogen system module can be installed or removed independently. The hydrogen system is arranged and located to provide access to the service propulsion engine.

Figure 91 also shows the accumulator (surge tank), heat exchangers, and flow restrictors in the CM oxygen supply system. These components are part of the ECS but serve important functions for the cryogenic storage subsystem. The heat exchanger, in conjunction with the flow restrictors, prevent the loss of system pressure in the cryogenic oxygen storage system



NOTE: THE EQUIPMENT LOCATED IN THE COMMAND MODULE IS NOT PART OF THIS SYSTEM. IT IS SHOWN FOR CLARITY ONLY.

Figure 91. Schematic Arrangement of the Apollo Block II Cryogenic Storage Subsystem

during emergency flow conditions in the event of a cabin puncture. The combined flow from the accumulator and the cryogenic storage system is required to provide the emergency flow for five minutes.

Flow from the reactant tanks to the fuel cells is controlled by solenoid valves, which in turn are controlled by pressure regulators in the fuel cell. Flow from the oxygen tanks to the ECS is controlled by flow restrictors and pressure regulators in the ECS.

The duration and power requirements of the extended Apollo missions affect the implementation of Apollo CGSS components in three major areas:

1. Service life

2. Fluid storage

3. ECS expulsion requirements

Present Apollo CGSS components have a minimum service life requirement of 400 hours. Service lives of approximately 900 hours are required for 30-day mission and 2300 hours for ninety-day mission. All components must be qualified to these criteria. Those components (i.e., solenoid valves, pressure switches, check valves) which undergo numerous cycling operations during their service life will present the biggest problems associated with qualification and acceptance testing. Other components (i.e., pressure relief valve, pressure transducer, temperature sensor) which do not experience excessive cycling operations may not prove difficult to qualify.

Fluid Storage Heating Requirements

The amount of fluid necessary for the mission is dependent on the energy demands on the EPS and the storage requirements of the ECS. The EPS demand is composed primarily of the amount required for experimentation (as defined by the customer), the parasitic power of the EPS itself, and the CGSS power requirements.

The CGSS energy requirements are composed primarily of the amounts necessary to operate the tank heaters and fan motors; nominal amounts are required for the pressure switch, solenoid valves, pressure transducers and temperature sensors.

For constant pressure operations, the required heating rate is

$$q_r = (q_s)\,\dot{w} - q_a$$

where

q_s = specific heat input, Btu/lb of fluid withdrawn

$\dot{w}$ = flow rate, lb/hr

q_a = allowable heat leak

The specific heat input is a function of the fluid thermodynamic properties. Flow rate is

$$\dot{w} = R \times E \times P$$

where

R = ratio, lb of fluid/lb of water (1/9 for hydrogen; 8/9 for oxygen)

E = powerplant efficiency factor, lb of water/kw-hr

P = power plant demand, kw-hr

Powerplant efficiency factors for the Apollo fuel cell vary from approximately 0.67 to 0.88 lb H_2O/kw-hr; 0.77 is used in the following calculations as a reasonable value for preliminary purposes.

To prevent system boil-off losses, the allowable heat leak is

$$q_a = q_{s_{(min)}} \times \dot{w}_{(min)}$$

For a power of 2.0 kw, the minimum flow rate is

$$= 0.77 \times 2.0 \times 1.0/9.0$$

$$= 0.171 \text{ lb/hr (for hydrogen)}$$

$$= 0.77 \times 2.0 \times 8.0/9$$

$$= 1.37 \text{ lb/hr (for oxygen)}$$

The allowable heat leak is:

$$= 0.171 \times 100 = 17.1 \text{ Btu/hr (hydrogen)}$$

$$= 1.37 \times 32 = 43.9 \text{ Btu/hr (oxygen)}$$

A reasonable estimate can be made for the heating requirements of the hydrogen system if it is assumed that the average required heating rate is 30 percent of the maximum heating rate.

$$q_{s\ (max)} = 262\ \text{Btu/lb}$$

$$\dot{w}_{(max)} = 0.77 \times 4.2 \times 1.0/9.0 = 0.36\ \text{lb/hr}$$

$$q_{r\ (max)} = 262.0 \times 0.36 - 17.1 = 77.8\ \text{Btu/hr} = 22.6\ \text{watts}$$

$$q_{r\ (max)} = 6.8\ \text{watts} = q_{av}$$

However, a reasonable estimate of the heating requirements of the oxygen tanks is difficult because of the expulsion requirements of the ECS. Heating requirements vary directly with the amount of fluid remaining in the tank. If ECS oxygen is stored cryogenically, it is necessary to know when the oxygen expulsion takes place in order to determine the heating rate.

The average heating rate for the EPS oxygen is 37.6 watts (assuming $q_{r(av)} = 0.3\ q_{r(max)}$. A summary of the estimated power requirements for the CGSS is shown below as a function of the number of pairs (O_2 and H_2) of reactant tanks. The heating requirements for the ECS must be added to the values shown below if the oxygen is stored cryogenically. The CGSS electric power and energy requirements are shown in Table 58.

In addition to the heat required for the oxygen tanks (which include ECS provisions) some heat may be required to increase the temperature of the oxygen delivered to the ECS. The principal reason for this requirement is the relatively high flow rates encountered during the various repressurizations. An investigation of this requirement is beyond the scope of this report

Table 58. CGSS Electric Power and Energy Requirements

No. of Pairs of Tanks	Tank Heaters (watts)	Fan Motors (watts)	Instruments (watts)	Subtotal (watts)	Energy Required (kw-hr)	Solenoids Valves (kw-hr)	Subtotal (kw-hr)	Losses (15%) (kw-hr)	Totals (kw-hr)
1	45	19	5	69	50.0	0.3	50.3	7.5	57.8
2	45	38	10	93	67.5	0.3	67.8	10.2	78.0
3	45	57	15	117	85.0	0.3	85.3	12.8	98.1
4	45	76	20	161	102.5	0.3	102.8	15.4	118.2

since it requires a knowledge of the ECS design, repressurization occurrence times, and initial temperature. Also, a trade-off study should be conducted to determine if the required heat may be more economically provided by a heat exchanger system as opposed to electrically provided energy.

Because of the relatively low flow rates associated with fuel cell operation, it is probable that no heat need be provided to the reactant gases prior to their entrance at the fuel cell pressure regulators. It is anticipated that the heat provided from the surrounding structure should be sufficient to satisfy the minimum temperature requirement at the fuel cell entrance. As before, a thorough heat-transfer analysis is necessary to verify this assumption.

Heat is required at the reactant tanks prior to launch in order to build up pressures to operating pressures. It is anticipated that this energy will be supplied by ground equipment. For the constant volume process, this amount of heat is

$$Q = W\Delta h - \frac{V\Delta P}{J}$$

It can be shown that this relationship becomes

$Q \approx 7.49\ W$ (Btu, oxygen system)

$\approx 13.20\ W$ (Btu, hydrogen system)

It should be noted that as the remaining weight in the reactant tanks approaches 10 percent of the initial weight, the specific heat input becomes asymptotic and q_r increases correspondingly, depending upon the withdrawal rate ($\dot{w}$). However, the temperature of the fluids becomes a controlling factor in the evaluation of the heating requirements towards the end of the mission. It is probable that the heaters must be turned off and the tank pressure allowed to decay in order to prevent the fluid temperatures from exceeding 70 F, the minimum allowable fluid temperature at the fuel cell. In order to verify this assumption, however, it would be necessary to conduct a transient heat-transfer analysis which would consider all the factors affecting the temperature of the stored gas.

Apollo Block II Cryogenic Gas Storage Tank Heat-Leak Analysis

The heat-leak characteristics of the Apollo Block II cryogenic gas storage vessels have been analyzed so that the electrical energy requirements for extended missions can be accurately determined.

The objective of the calculations was to determine the heat leak into the Apollo cryogenic tanks during varying flow-rate conditions where the flow is higher than the minimum normal flow rate. Heretofore the only values available concerning the heat leak of the Apollo Block II cryogenic storage tanks have been for conditions of minimum flow throughout the complete density spectrum. When the flow rate from the vessel is increased above the minimum level, the heat leak through the insulation to the inner vessel is decreased because of the increased cooling of the shield and insulation.

The results of the analysis are summarized in Figures 92 through 95 and present the heat input requirements as a function of flow rate for oxygen and hydrogen. Figures 92 and 93 are for high and low density oxygen, respectively, and Figures 94 and 95 are for high and low density hydrogen, respectively.

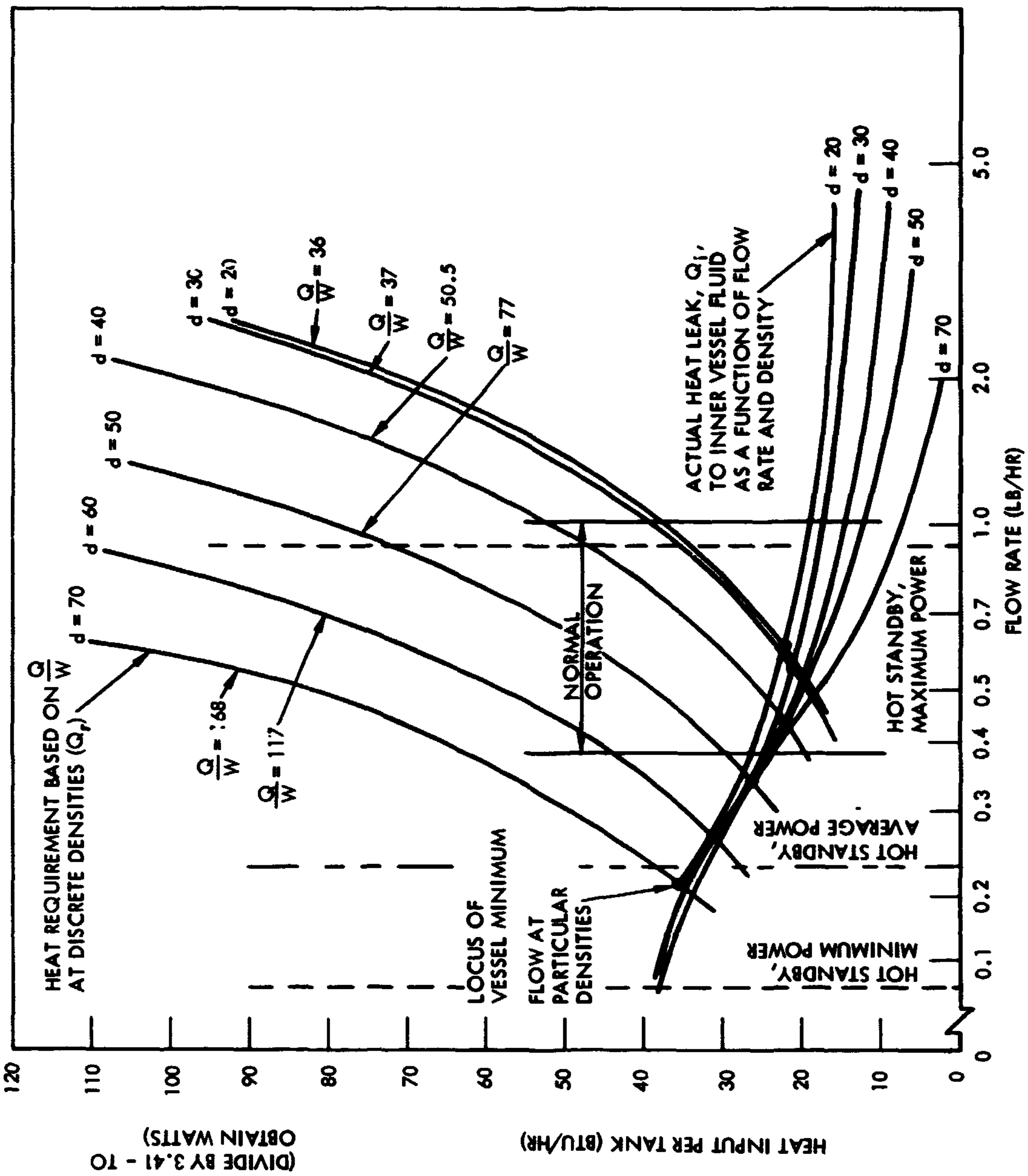
The horizontal parameters of the figures relate to the heat leak into the fluid as a function of density and flow rate. Vertical parameters relate to the total heat input requirements as a function of density and flow rate. The difference between the two curves on Figures 92 through 95 is the electrical power required to maintain operating pressure level.

These curves show that the heat leak into the inner fluid decreases as the flow rate increases due to the decreasing shield temperature. The heat leak also varies with density in that it increases with decreasing density until the min dc/dm density is reached, and then the heat leak decreases due to increasing fluid temperature.

The analysis is based on the standby heat leak of the tanks at zero flow, and on the vapor cooled shield performance during flow. A heat balance across the insulation system was obtained based on the heat absorption capabilities of the exiting fluid. As the flow rate increased, then the heat abroption of the shield increased, which caused the shield temperature to decrease, even though the heat leak from the outer shell to the vapor cooled shield increased.

Constraints on Reactant Consumption Rates Due to Thermal Build-up of Pressure in CGSS Tanks

Tests at SID on Block II CGSS tanks have established flow rates of hydrogen and oxygen at the respective minimum dQ/dM or worst-case condition. Heat leak into the cryogenic fluid at the minimum dQ/dM is the lowest rate of heat leak possible during a mission which is required to expel a unit mass velocity of fluid. Units of dQ/dM are, typically, Btu/pound. The test results are given in Table 59.



TO USE CURVE:

1. SELECT FLOW RATE (W)
2. OBTAIN HEAT LEAK INPUT AT PARTICULAR FLUID DENSITY Q_i
3. OBTAIN TOTAL FLUID ENERGY REQUIREMENTS, QR, AT PARTICULAR FLUID DENSITY, AT W IN QUESTION
4. DIFFERENCE OF $(Q_r - Q_i)$ IS HEAT ADDITION THAT MUST BE SUPPLIED TO STORED FLUID, TO MAINTAIN PRESSURE
5. NOTE: HEAT INPUT BY HEATERS AND FANS IS 580 BTU/HR PER TANK

Figure 92. Oxygen Heater Input Requirements, High-Density Region

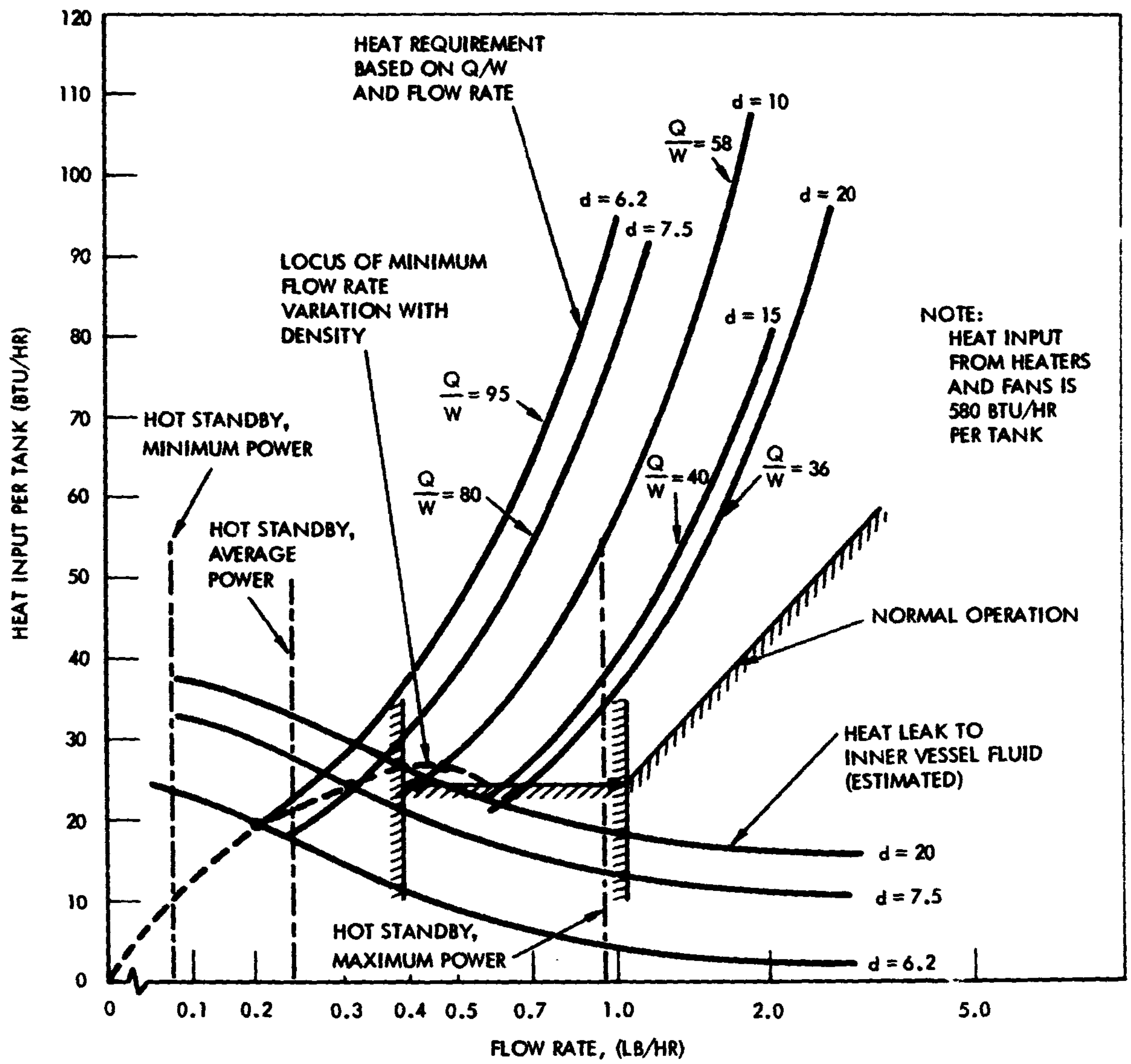


Figure 93. Oxygen Heater Input Requirements, Low-Density Region

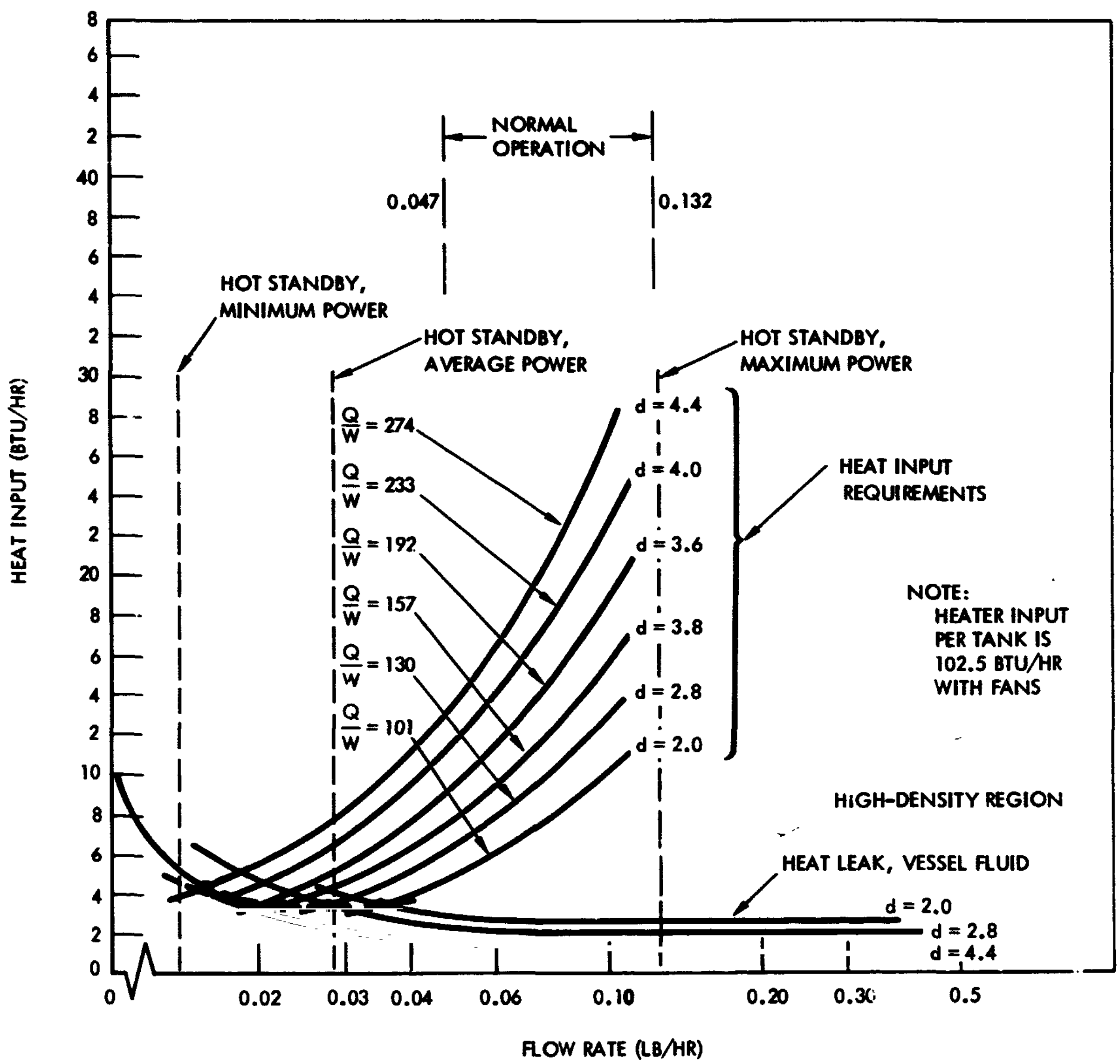


Figure 94. Hydrogen Heater Input Requirements, High-Density Region

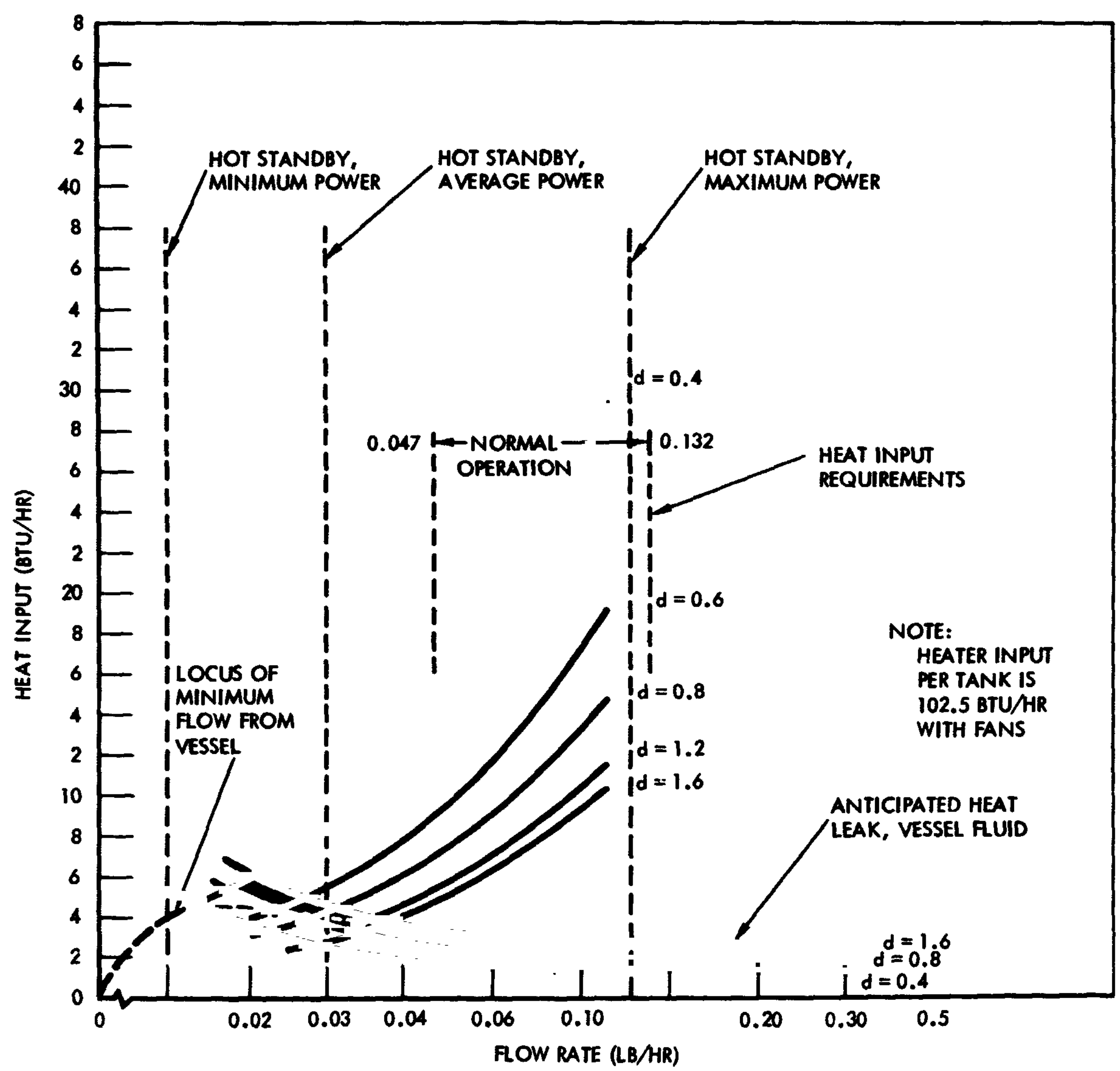


Figure 95. Hydrogen Heater Input Requirements, Low-Density, Region

Table 59. CGSS Test Data of Minimum Fluid Flow Rates for Minimum dQ/dM Condition - Ambient Temperature 140 F

Hydrogen Tank No.	Flow Rate (lb/hr)	Oxygen Tank No.	Flow Rate (lb/hr)
1	0.0660	1	0.667
2	0.0730	2	0.690
3	0.0713	3	0.678
4	0.0709	4	0.713
5	0.0720	5	0.707
		6	0.712
		7	0.667
		8	0.693
Present design goal	0.054	Average	0.691 ± 0.015

From parameters of Table 59 (namely, the design goal worst-case hydrogen flow rate and average test value of worst-case oxygen flow rate, which pertain to 140 F local ambient), flow-rate parameters were computed for a 30 F local ambient environment which is believed to represent an effective heat-sink temperature for the RCM-installed EPS gas supply. These parameters relate to both a worst-case condition and an average condition which is based upon the simplifying assumption that, for an entire mission, a thermally required average flow rate is approximately 0.8 times the worst-case flow rate. These parameters are combined with flow rates required by fuel-cell operation to form Table 60.

Control of CGSS Tank Temperature

The Block II cryogenic tanks must be maintained at the proper temperature to minimize venting the reactants overboard. To accomplish this, AAP studies have recommended adding an insulation shield around the tanks. The type shield recommended for the AAP is as follows.

Table 60. Two-Pair Tank CGSS-EPS. Fluid Flow Parameters Under Normal and Limiting Conditions

	Hydrogen	Oxygen
Average flow rate for 30-Day mission, lb/hr tank	0.0389	0.311
Range of specific reactant consumption, lb/kw hr	0.0744 - 0.0977	0.595 - 0.784
Most probable average specific reactant consumption	0.0855	0.684
Flow rate for minimum dQ/dM condition, lb/hr tank	(ambient 140 F) 0.054 (ambient 30 F) 0.049	(140 F)(EPS+ECS) 0.691 (30 F)(EPS+ECS) 0.545 (140 F)(EPS) 0.485 (30 F)(EPS) 0.382
Estimated average minimum flow rate requirement based on average dQ/dM condition	(ambient 140 F) 0.043 (ambient 30 F) 0.039	(140 F)(EPS+ECS) 0.560 (30 F)(EPS+ECS) 0.436 (140 F)(EPS) 0.388 (30 F)(EPS) 0.306
Hot standby flow rate,* lb/hr tank — minimum	0.008	0.064
average	0.028	0.204
maximum	0.117	0.936
Normal operating flow rate, lb/hr tank — minimum	0.024	0.192
average	0.0389	0.311
maximum	0.061	0.488

0.563 kw, minimum power level - normal operating mode
1.428 kw, minimum power level - normal operating mode

*Assumes one fuel cell is operating normally, one is on hot standby.

Superinsulation comprising 40 layers of crinkled aluminum-coated Mylar sheet material, forming a gross thickness of approximately 0.5 inch, will be placed midway between two aluminum sheet metal bumper shields which are set approximately 1.0 inch apart. Each bumper shield is 0.016 or 0.020 inch thick, depending upon configuration size. Space between the two bumper shields is space-vacuum, with the superinsulation occupying the mid-region. Walls formed in this manner are then arranged and attached together to form a continuous enclosure around a group of hydrogen tanks, and a group of oxygen tanks.

If, for a particular mission, a two-pair gas supply system is to be installed, the two hydrogen tanks will be installed together and totally enclosed, (except for the base support structure) by the bumper-shield-superinsulation structure. The two oxygen tanks will be similarly enclosed.

To interrupt or reduce heat leak into each cryogenic tank, support structure attachments will be fabricated from low thermal conductivity ceramic material. A design based upon this technique can be expected to minimize heat leak into the supercritical fluid. An internal effective ambient temperature around the tanks of 30 F has been set as a preliminary design goal. This applies to both hydrogen and oxygen tanks when the external environment temperature is no greater than 140 F. Again, a specific design recommendation must be withheld until definition of a mission and establishment of the actual environmental conditions to which the CGSS will be exposed.

CGSS - EPS Configuration

From the fluid-flow parameters listed in Table 60, it is seen that a single pair of tanks with either one or two fuel cells is feasible. Likewise, two pairs of tanks with either one or two fuel cells are a feasible configuration for a 30-day mission. In either case, a peaking battery normally would supply peak power.

Analysis and Description of Present Block II Radiator

The present Apollo Block II radiator area of 40 square feet will be used as the primary heat-rejection mechanism for the AES fuel-cell waste heat. The present configuration can be changed to accommodate the addition of a fourth fuel cell. There are presently eight individual panels of approximately 5 square feet each (i.e., 20-1/4 by 36 inches), with three tubes on a panel corresponding to one fuel cell each (one parallel passage per fuel cell).

The heat transfer behavior of a number of radiator configurations was analyzed. The Block II radiator capability under worst conditions is taken to be 5499 Btu/hour (See Table 61).

Table 61. Apollo Block II Capability (40 Square Feet, Lunar Orbit (80 Miles) Worst Orientation)

Section of Eight-Sectioned Circumference	Q_{in} $\frac{\text{Btu}}{\text{Hr-Ft}^2}$	Qreject $\frac{\text{Btu}}{\text{Hr}}$ (Net)	Specifications: Per Module
1	159	242	T_{in} = 695 R
2	87.2	357	T_{out} = 668 R
3	73.7	362	W = 83 lb/hr
4	72.4	313	Q_{reject} = 1833 Btu/hr (per module)
5	89.5	320	Q_{reject} (3 modules) 5499 Btu/hr
6	165.6	204	
7	304	13	
8	318	-5	

Net heat rejection varies along the circumference of the vehicle as a result of different incident heat loads. The lower net rejection for sides 1, 6, 7 and 8 results from high incident energy from lunar planetary thermal emission. The most critical condition for the radiator is when it sees the lunar surface, since selective coatings are not effective for the longer wavelengths of the thermal planetary emissions. Side 4 (opposite side 8, the worst side) is therefore selected as the required radiator location for any additional area. Under this restraint, the radiator is expected to result in a net heat rejection of 208 Btu/hour per ft^2 of effective area. Additional area requirements can thus be determined. For example, for a heat-rejection rate of 8500 Btu/hr, corresponding to 4000 watts of power output, the required area increase is: (1) 40 square feet results in 5500 Btu/hour net heat rejection, (2) excess waste heat = 8500 - 5500 = 3000 Btu/hour, and (3) area to be added = 3000/208 14.4 square feet.

If the radiator operating temperatures and conditions are lowered (T_{in} = 643 R, T_{out} = 608 R, and W = 78.5 pounds per hour), the net heat rejection will reduce to 145 Btu/hour-ft^2 and the required additional area becomes 20.7 square feet.

To increase fuel cell output to the maximum capability of 4000 watts (i.e., two FCP operating at 2000 watts each), the radiator area must be increased accordingly. For minimum load conditions, the maximum active radiator area is limited to 48 square feet; i.e., a minimum heat rejection of 2550 Btu/hour results in an equilibrium temperature of -30 F (taken to be the lower limit for water glycol) with an active area of 48 square feet.

During space coast of the mission, with the vehicle facing the sun and spinning, the net heat-rejection capability of the full 40-square foot area is taken to be 6810 Btu/hour (see Table 62), which is adequate for the estimated requirements of the fuel cells. This value represents steady-state operation, with a radiator inlet temperature of 643 R and an outlet temperature of 608 R at a flow rate of 78.5 pounds per hour.

Apollo Block II Radiator Capability

The following Block II radiator capability is based upon the analysis performed during this study. For earth orbit, a minimum heat rejection of 1746 Btu/hour is required to maintain 40 square feet of radiator surface (see Table 63) above -35 F as a radiator outlet temperature. This is taken to be the lower limit for the water-glycol mixture that is used as the coolant. With a 200 F condenser temperature as maximum flowable, the radiator heat-rejection capability is 7270 Btu/hour. Figure 96 shows the effect of varying this condenser exit temperature (i.e., varying radiator surface temperature). Two ratios of thermal coatings are considered with α/ϵ = 0.18/0.92 as the initial value and α/ϵ = 0.27/0.85 as a final value. Degradation of surface coating has not been fully defined for all space environment (i.e., low-energy proton fluxes); however, α/ϵ = 0.27/0.85 is taken as the maximum degradation of coating.

Figures 97 and 98 show maximum and minimum heat-rejection values as a function of gross power for Block II. Figure 99 gives the glycol pump characteristics, with curve No. 1 representing maximum values and curve

Table 62. Block II Capability - Space Coast (Facing Sun and Spinning)

Side	$Q_{in} \frac{(Btu)}{Hr\text{-}Ft^2}$	$Q_{reject} \frac{(Btu)}{Hr}$	One Module
1	0	322	T_{in} = 643 R
2	0	355	T_{out} = 608 R
3	0	343	W = 78.5 lb/hr
4	0	293	Q_{reject} = 2270 Btu/hr (per module)
5	53	252	Total Q_{reject} 2270 (3) = 6810 Btu/hr
6	73	218	
7	73	186	
8	20	280	

No. 2 minimum values. A slight increase in net heat rejection is obtained by using curve No. 2 for values shown in Figure 96 (e.g., approximately 30 Btu/hour at maximum condenser temperature of 200 F). Table 63 shows Block II capability for lunar-transit phase. Minimum heat rejection at minimum incident energy (that is, Q_{in} = 0) is shown to be 2457 Btu/hour. Below this, heat-rejection rate radiator fluid freezing becomes a serious problem for the condition of deep-space or zero incident heat.

Utilization of Apollo Block II Radiator Panels

Eight Apollo Block II radiator panels comprising an area of 40 square feet are designed to reject heat from the Block II three-fuel cell system.

Table 63. Block II Radiator Comparisons

100-Nautical Mile Earth Orbit X-Axis Horizontal (Area = 40 Square Feet)				
Glycol Pump*	1	2	1	2
Condenser maximum exit temperature (F)	200	200	**	**
T_{in} (F)	214.3	223.4	10.7	11.5
T_{out} (F)	177.9	170.8	-13.7	-35.1
Loop ΔP lb/in^2	1.4	1.0	19.7	14.0
Flow lb/hr	81.6	56.6	35.5	17
Q_{reject} Btu/hr (net)	7270	7290	1746	1581

*Glycol pump characteristics relate to maximum and minimum pressure drops per Figure.

**Minimum condenser exit = -35 F radiator outlet

Table 63. Block II Radiator Comparisons (Cont)

Transit Phase of Mission										
	X-Axis ⊥ Vehicle Sun Line		X-Axis ∥ Vehicle* Sun Line		X-Axis ⊥ Vehicle Sun Line 5/8 Area		X-Axis ∥ Vehicle Sun Line 5/8 Area		X-Axis 20° Off ⊥ Vehicle Sun Line 5/8 Area	
Glycol Pump***	1	2	1	2	1	2	1	2	1	2
Condenser maximum exit temp. F	170	170	**	**	170	170	**	**	**	**
T_{in} (F)	183.8	192.4	9.6	34.1	177.5	182.7	16.4	9.5	5.8	11.9
T_{out} (F)	147.7	141.0	-34.9	-34.9	156.5	152.4	-5.4	-34.9	-20.2	-35.0
Loop ΔP lb/in.^{2}	2.0	1.4	25.4	13.3	1.6	1.1	15.6	13.6	21.3	13.6
Flow lb/hr	78.6	55.3	27.7	19.3	81.0	56.5	41.4	18.2	33.4	18.4
Q_{reject} Btu/hr (net)	6813	6825	2457	2701	4119	4128	1842	1617	1743	1716

*Representative of lunar-orbit minimum heat-rejection conditions
**Minimum condenser exit = -35 F radiator outlet
***Glycol pump characteristics relate to maximum and minimum pressure drops per Figure.

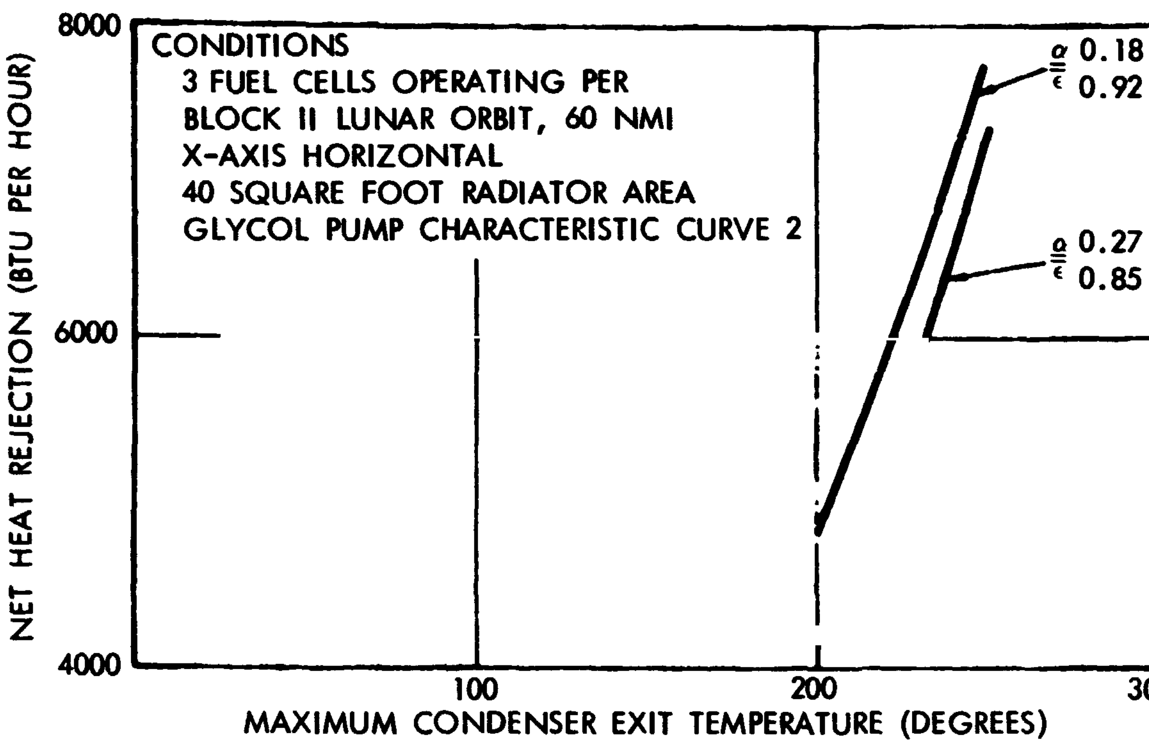


Figure 96. Apollo Block II Radiator Capability, Lunar Orbit

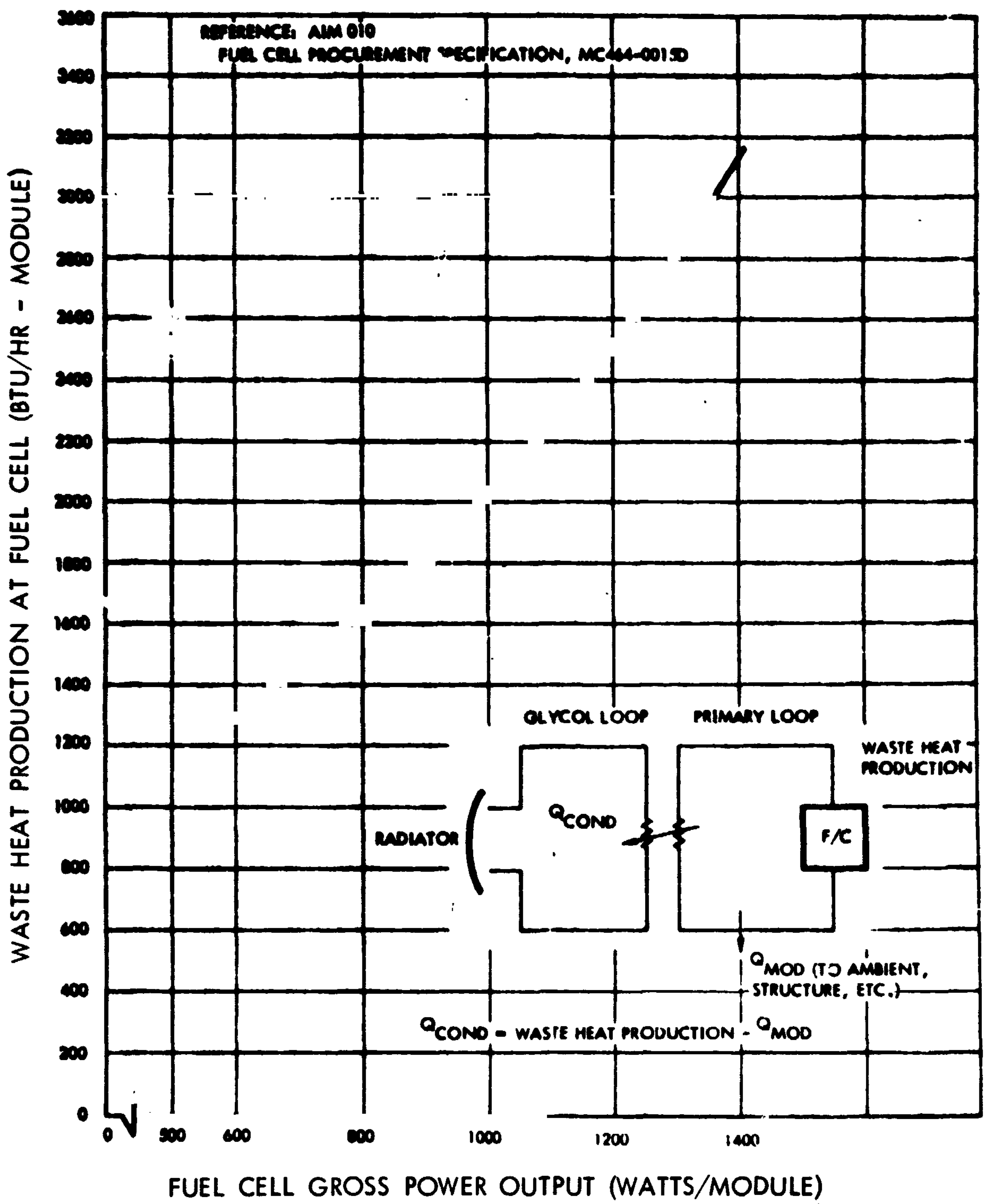


Figure 97. Minimum Fuel-Cell Heat-Production Rate

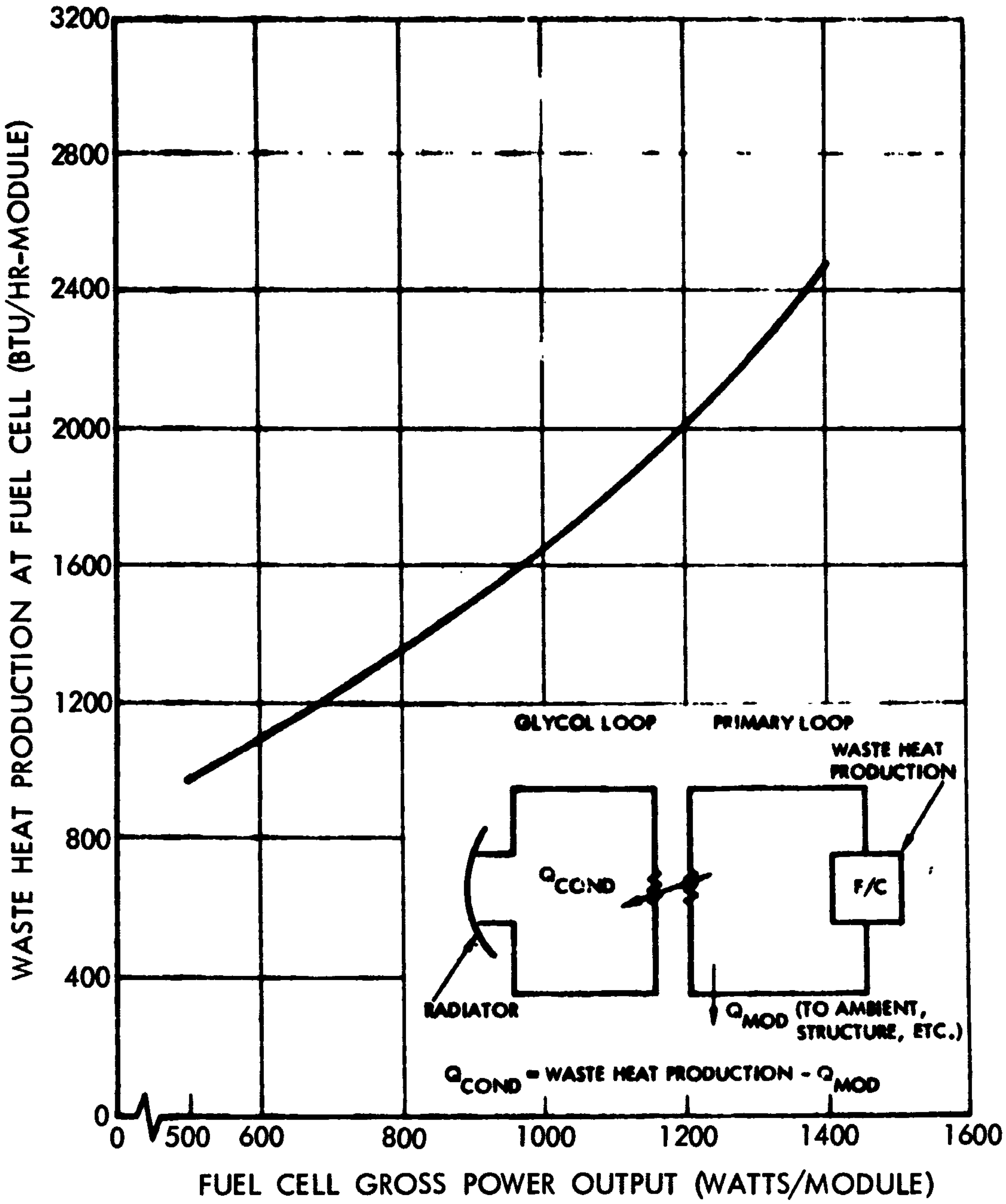


Figure 98. Minimum Fuel-Cell Heat-Production Rate

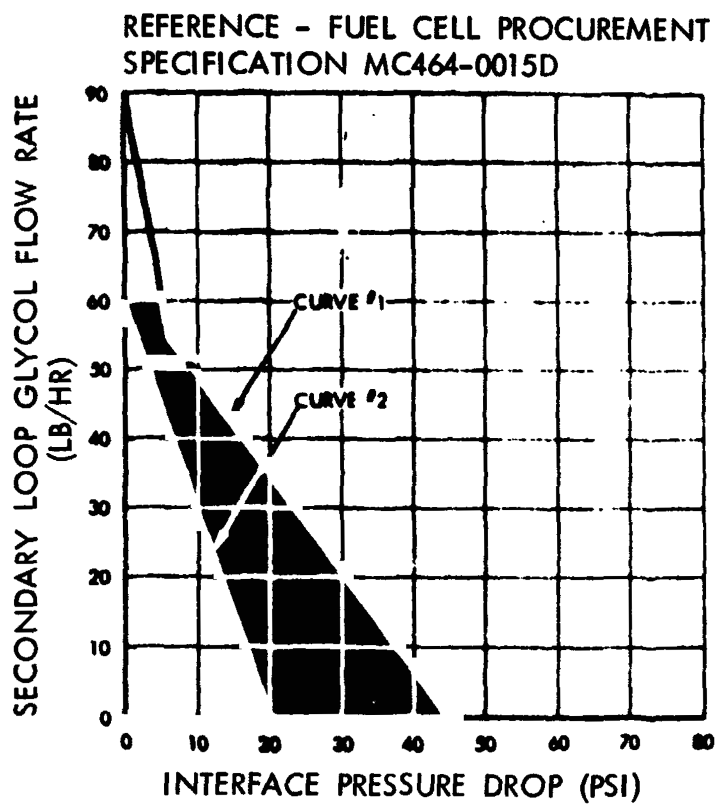


Figure 99. Glycol Pump Characteristics

If, for a particular mission as yet undefined, a one-fuel cell system is required, then a proportionate number of Apollo radiator panels would be installed (i. e., three panels). Likewise, two fuel cells require two-thirds of a full set (i. e., five or six panels depending on the mission).

As previously described, each Block II radiator panel includes three fluid coolant tubes arranged in geometric parallel. In the three-fuel cell Apollo system, each coolant tube relates to a particular fuel cell.

If a three-fuel cell system is to be installed for the RCM laboratory, the Apollo Block II system installation is to be followed. Plumbing interconnections from panel to panel follow a regular sequential order. The tube of fuel cell No. 1, which is tube No. 1 (counting from top) on panel No. 4, takes the position of tube No. 2 (second from top) on panel No. 5, and so on. This arrangement reduces side effects in heat radiation to space.

If a one-fuel-cell system is to be installed for the laboratory, several methods of using the available tubes are possible. They can be connected in mechanical parallel, mechanical series, or some combination of parallel and series. If in series, the tubes can be arranged in either the forward- or counter-flow direction. Even if the tubes are series-connected in forward-flow arrangement, however, they are heat-transfer countercurrent, since there is a difference in fluid temperature in adjacent tubes at points of entrance. The main advantage of series arrangement is to use each radiator panel, effectively, as a type of auxiliary heat exchanger. During periods of maximum heat rejection, the radiator is somewhat less effective than a parallel-flow type. However, during periods of minimum heat rejection with the panel pointed towards deep space, coolant is less likely to freeze since some heat is picked up from an adjacent tube. Parallel arrangement has two important advantages: (1) greater heat rejection is possible, and (2) less fluid friction in the coolant loop requires less work by the pump, resulting in longer service life and less electric power input. Therefore, the parallel arrangement, with the necessary plumbing line changes and manifolding (only two in each loop), is recommended for the RCM laboratory EPS-radiator design. Addition of a 5-watt electric resistance heater, to bypass lines of each radiator cooling loop and possibly to direct lines as well, might very well provide positive control of the physical condition of coolant fluid by eliminating, in a practical sense, the possibility of freezing.

Subsystem Buildup

Selection of a power source is based largely upon the energy storage required for the mission. The dependent system could only be used for relatively low-level operation unless the CSM has been modified to increase its cyrogenic storage capacity. Figure 100 shows a comparison of the energy capability of batteries and fuel cells on a weight basis. The battery data are based upon the batteries listed in Table 42, and the fuel cell curve is based upon current operational data and does not include the low-level operation previously described. The superiority of one source over the other is definite at the ends of the curve. But the gross nature of the calculations limits its usefulness at the crossover point. If the energy levels fall in the 30-kw-hour range, further study is required before any selection should be made.

After the power source is selected, the rest of the EPS can be built up from the existing Apollo components. Tables 64 and 65, list the physical characteristics of the Block I and Block II components previously listed in Table 19. Each component's function and capacity is listed in Table 66. A guide to the application of these components is given in Table 67, EPS Subsystem Buildup.

An example of the use of these tables is given herein:

1.	Power requirement	Less than 200 watts with short peaks of 1000 watts
	Selected source	Dependent d-c power plus peaking battery
2.	Support requirements	IIa and IIc call for Ia, IVb which call for IIIa (note 1) which yields IVa.
3.	EPS components	a. LEM docking umbilical
		b. C14A16
		c. C14A2
		d. C14A7
		e. C15A5
		f. C14A3

Note 1: An independent a-c source was selected to minimize the impact to the CSM wiring.

POWER SOURCES

DEPENDENT POWER FROM CSM

<200 WATTS AVG; <1000 WATTS PEAK

BATTERIES VS FUEL CELLS

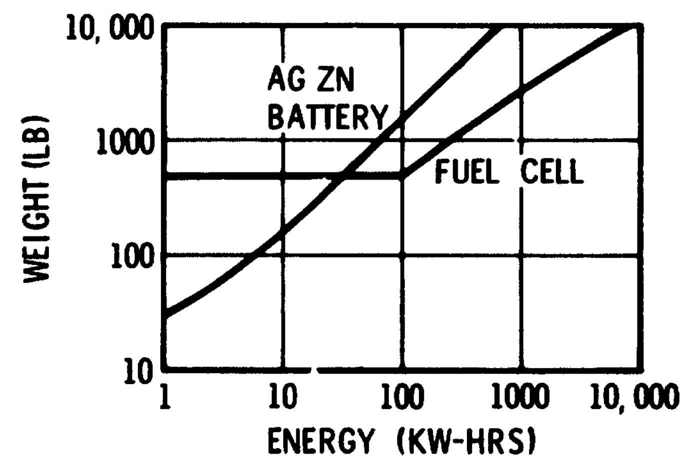
- BATTERIES

  0.75 TO 440 AMP-HR
  SIZES AVAILABLE

- APOLLO FUEL CELL

  900 WATTS AVG/MODULE
  FOR 30 DAY MISSION

  1375 WATTS/MOD MAX



SOLAR CELLS

- RELATIVELY EXPENSIVE
- CONSTRAINTS: ORIENTATION; RAD AND METEOR. DEGRAD; THERMAL CONTROL;
  ECLIPSE PROVISIONS
- PROVIDES LONG TERM OPERATION

Figure 100. RCM Laboratory EPS

Table 64. Block I Component Physical Characteristics

Component	Dimension (in.) X	Y	Z	Volume ($in.^3$)	Weight (lb)	Temperature (F) Low	High	Coldplate
C14A1	47	72	17.4	590	10.6	-55	200	No
C14A2	62	91	103	580	7.6			
C14A3	55	8.6	2-9	260	80			
C14A7	6.8	41	16.2	452	5.2			
C14A8					4			
C14A12	65	4	35	91	4			
C14A13	4	118	71	335	7			
C14A14	152	20.2	20-8.4	4360	20			
C14A15	94	10.8	5	508	70			
C14A16	6	10.6	6-12.2	580	3.4	-55	200	No
C14BT 1, 2, 3	6.9	5.75	11.75	4.66	28	50	200	Yes
C14BT 5, 6	3	2.75	6.75	56	5.6	50	200	Yes
C14BTC 1	3	6.3	5.7	108	3.8	50	200	Yes
C14PS 1, 2, 3	69	146	149	1490	39.5	-55	200	Yes
BVT		Tubing only		Negligible	1.5	-	-	-
C15A 1					3.0	-65	140	No
C15A2					3.0			
C15A5	85	4	95	323	70			
S14A1					35.3			
S14A2					5.5			
S14PS 1, 2, 3	44	22.5 diameter cylinder		17500	262			
S48A4, 5					2			
S48A6					25			
S48A7					25			
S48A10, 11					7.1			
S48AR 107, 108, 109, 110, 111, 112					1			
S48LV1, 2, 3, 4, 5, 6					4	-65	140	No
S48TK 1, 2					70	-55	140	No
S48TK 2, 3					85	-65	140	No

Table 65. Block II Component Physical Characteristics

Component	Dimension (in.)			Volume	Weight	Temperature (F)		Coldplate
	X	Y	Z	($in.^3$)	(lb)	Low	High	
C14A1	4.5	8	14.5	522	10.6	-55	200	No
C14A2	5.4	9.25	12	600	7.6	-55	200	No
C14A3	47	122	75	430	80	-55	200	No
C14A7	6.2	4	16	398	5.2	-55	200	No
C14A8	42	108	76	378	4	-55	200	No
C14A12	6.5	4	3.5	91	4	-55	200	No
C14A13	4.25	12.2	12.2-6.2	472	100	-55	200	No
C14A14	-	-	-	-	-	-55	200	No
C14A15	-	-	-	-	-	-55	200	No
C14A16	13.5	5.2	2.8	197	3.4	-55	200	No
C14BT 1, 2, 3	6.9	5.75	11.75	466	28	50	200	Yes
C14BT 5, 6	3	275	6.75	56	5.6	50	200	Yes
C14BTC 1	3	6.3	5.7	107	3.8	50	200	Yes
C14PS 1, 2, 3	6.8	14.8	14.8	1490	39.5	-55	200	Yes
BVT		Tubing only		Negligible	1.5	-	-	-
C15A 1	14	10	3	420	6	-65	140	No
C15A 2	10	3	10	300	4			
C15A 5	8.4	50	9.1	306	70			
S14A 1	7.6	20.8	184	2910	35.3			
S14A 2	4.9	7.7	108	411	5.5			
S14PS1, 2, 3	44	22.5 diameter cylinder		17,500	265			
S48A4, 5	25	4	4	40	2			
S48A6	5	10	6	300	12			
S48A7	5	10	6	300	12			
S48A10, 11	54	51	51	141	7.1			
S48AR 107, 108, 109, 110, 111, 112	3	3 diameter cylinder		21	1			
S48LV 1, 2, 3, 4, 5, 6	2.3	5.5	4.5	57	4	-65	200	No
S48TK 1, 2	33 diameter sphere			18,800	70	-65	140	No
S48TK 3, 4	21.5	28.2 diameter cylinder		13,300	82	-65	140	No

Table 66. EPS Component Functions

Component	Function	Capability
C14A1	Selects a-c source and connects to bus plus a-c sense unit	Selects 1 of 3 sources
C14A2	Battery bus tie	2 batteries
C14A3	Inverter d-c control	3 inverters
C14A7	Bus tie and inverter circuit breakers plus isolation diodes	13 circuit breakers + 8 diodes
C14A8	Fuel-cell pump motor phase correction	3 fuel cells
C14A12	Circuit protection	30 fuses
C14A13	Control uprighting system compressor	3 motor switches and 2 circuit breakers
C14A14	Control auxiliary battery supply on BL I	Control 3 batteries
C14A15	Circuit breaker for auxiliary power supply	7 circuit breakers
C14A16	Battery and pyro circuit breakers	9 circuit breakers
C14BT1, 2, 3	Postlanding and reentry batteries	40 AH at 27 volts
C14BT5, 6	Pyrotechnic batteries	45 A min at 23 volts
C14BTC1	Battery charger	1 battery at a time
C14PS1, 2, 3	D-c to 3ϕ a-c converter	1250 VA
	Vent battery gasses overboard	3 batteries
C15A1	Bulkhead feedthrough and pyro disconnect	9 movable, 3 fixed connectors
C15A2	Bulkhead feedthrough and pyro disconnect	5 movable, 1 fixed connector
C15A5	Voltage sensor	2 busses
S14A1	Controls fuel-cell output	3 fuel cells to 2 busses
S14A2	Controls fuel-cell operation	3 fuel cells
S14PS1, 2, 3	Converts H_2 and 0_2 to d-c power	900 watts
S48A4, 5	Conditions tank temperature and level sensors	1 tank
S48A6	Cryogenic H_2 relief and check valve for tank	2 tanks
S48A7	Cryogenic 0_2 relief and check valve for tank	2 tanks
S48A10, 11	Controls cryogenic tank motors and heaters	2 tanks
S48AR107, 108, 109, 110, 111, 112	Monitors FC gas flow rates	1 per reactant line
S48LV1, 2, 3, 4, 5, 6	FC gas shutoff valves	1 gas for 3 fuel cells
S48TK1, 2	Cryogenic H_2 storage	28 lb H_2
S48TK3, 4	Cryogenic 0_2 storage	320 lb 0_2
-	Support mounting skirt for H_2 tank	1 per tank

Table 67. E'S Subsystem Buildup

I. DISTRIBUTION

A. D-C .n

1. Support requirements (power source (II)

2. Components

a. C14A2 d-c power control
b. C14A7 main CB panel
c. C15A5 circuit utilization box
d. C152A05 panel, five controls
e. C22-2A203 displays

B. A-C System

1. Support requirements (power source (III)

2. Components

a. C14A1 a-c power control
b. C15-4A403 circuit breakers
c. C22-2A203 controls and displays

II. D-C POWER SOURCES

A. Dependent source

1. Support requirements (d-c distribution system (IA)

2. Components (LEM docking umbilical cables)

B. Primary battery

1. Support requirements

a. D-c distribution system (IA)
b. Coldplate (for C14BT 5, 6)

2. Components (C14A16 battery CB panel)

Table 67. EPS Subsystem Buildup (Cont)

C. Secondary Battery

1. Support requirements

a. D-c distribution system (IA)
b. Battery charger (IVb)
c. Coldplate (for C14BT 1, 2, 3)

2. Component (C14A16 battery 1 CB panel)

D. Fuel Cell

1. Support requirements

a. D-c distribution system (IA)
b. 3 ϕ a-c power source (III)
c. Gas supply (V)
d. Water glycol radiator and manifold (VI)

2. Components

a. S14A1 power distribution box
b. S14A2 F/C remote control assembly
c. S48A4197-112 flowmeter
d. S48LV1-6 F/C shutoff valves
e. F/C filter (ME286-0036)
f. C15-2A205 controls
g. C22-2A203 controls and displays

III. A-C POWER SOURCES

A. Dependent Source

1. Support requirements

a. A-c distribution system (IB)

2. Component (LEM docking umbilical)

B. Independent Source (Support Requirement)

1. Conversion equipment (IVA)

2. A-c distribution system

Table 67. EPS Subsystem Buildup (Cont)

IV. CONVERSION EQUIPMENT

A. Inverter

1. Support requirements

a. D-c power source (II)
b. Coldplate (for C14PS 1, 2, 3)

2. Components

a. C14A3 motor switch control
b. C22-2A203 control panel

B. Battery Charger

1. Support requirements

a. D-c power source (II)
b. 3 ϕ a-c power source (III)
c. Coldplate (for C14BTC 1)
d. 40 AHr secondary silver-zinc battery

2. Components

a. C22-2A203 panel 3 selector switch
b. C15A205 a-c power control

V. GAS STORAGE SYSTEM

A. H_2 Cryogenic Storage (S48TK 1, 2)

1. Support requirements

a. D-c power (II)
b. A-c power (III)

2. Component

a. S48AG valve module assembly
b. S48A10 cryogenic control system
c. Tank support skirt (ME127-0039)

Table 67. EPS Subsystem Buildup (Cont)

B. O_2 Cryogenic Storage (S48TK 3, 4)

1. Support requirement

   a. D-c power source (II)
   b. A-c power source (III)

2. Components

   a. S48A7 valve module assembly
   b. S48A11 cryogenic control system
   c. S48A4, 5 signal conditioner
   d. C22-2A202 control and display panel

VI. EPS RADIATOR

A. Support Requirements

1. Fuel cell (II-D)
2. Manifold assembly

B. Components

1. Space radiator ME901-0328-0002
2. Valve assembly ME284-0319-0001

The flexibility of the EPS control components can be attributed to their basically simple design. In many cases, the component assembly is just a mounting platform for the desired device, such as a circuit breaker, motor switch, or diode. The electrical terminals of these devices are brought to a terminal strip or connector on the assembly. The circuits are then completed by the external wiring harness.

This design concept makes it feasible to consider the possibility of utilizing these components is designs other than the one for which they were intended. For instance, the C14A2 d-c power control box consists only of two motor switches and four diodes. If a heavy-duty relay is required for some purpose, this unit probably would be able to perform the function. This type of philosophy, of course, depends upon the availability of the component for use. If the device is being used in its original circuit, another unit must be obtained and a new support structure provided. This may possibly exceed the cost of designing a new unit. Thus, the utilization of these components outside the original system must be judiciously considered in each case.

Solar Cell Power Systems.

Solar cells offer an attractive alternate power source for the missions of longer duration. Currently, a study is being made on the Apollo Application Program regarding the application of solar panels to the Apollo. Solar arrays have proven their capability on numerous spacecraft from the Vanguard to the Mariner programs and arrays as large as 400 watts have been flight tested on the OGO program.

The utilization of solar arrays incurs many mission constraints, however, as well as control problems that do not occur with the power systems previously discussed. The temperature of the panels must be held low, for example, to maintain the conversion efficiency of the cells. The N/P cells presently being used are relatively sensitive to nuclear radiation and protective devices, such as glass cover plates, and the avoidance of radiation belts need employment in order to increase the cells' useful lifetime. The angle of incidence of solar radiation and the earth's albedo also must be considered and methods of heliotroping the arrays must be employed. Some methods which have been used for this purpose are:

1. Panel orientation

2. Spacecraft orientation

3. Mounting cells over 360 degrees of surface

4. A combination of two or more of the above methods.

Furthermore, power must be maintained during eclipse periods and peak loads, hence a secondary battery system must be provided.

The advantages of the solar panels lie in their inherent simplicity, exclusive of the orientation problem, long life, and reliability. The output of the panels can be estimated to be eight watts per square foot for earth-orbit missions. If the panels are eclipsed for any appreciable time, the output of the system will have to be increased by as must as a factor of two to allow for battery charging. Current estimates of the price of the solar panels have been approximately $5000 per square foot. Thus, it can be seen that an array approximately the output of a fuel cell (1200 watts) cost approximately four times as must as the fuel cell.

No attempt has been made to design a solar cell system for the reason that, primarily, it is believed that one is beyond the scope of this report, and additionally, the lack of a specific mission would make any attempt in this direction relatively meaningless.

COMMUNICATIONS AND DATA

The Comm/Data building blocks are intended to provide various levels of capability for a partially dependent laboratory operation. This is, operation is possible only with the aid and use of a supporting CSM and its subsystems for certain required functions. This section presents three levels of capabilities provided by each configuration complement. Comm/Data Building Block No. 1 consists of a two-wire (exclusive of power requirements) hardline and an Apollo CM audio center. It is indicated part of the RCM Laboratory Comm/Data System in Figure 100. The hard line extends from the audio center (intercom bus) of a supporting CSM through the docking tunnel and into the audio center located in the Laboratory module. This configuration permits laboratory personnel to converse with each other or with the supporting CSM crew. The CSM can record voice on the DSE from laboratory crewmen for later dump via the CSM unified S-band equipment. S-band voice from the ground will be available to the laboratory, but the laboratory crew cannot transmit voice directly.

The audio center equipment is manufactured by Collins Radio Company and requires little or no refurbishment from previous flights. Physical characteristics of the audio center are: Weight 10 pounds; 5.8H x 10.3W x 7.0D (inches); power consumption is 20 watts. The audio center will require cold-plate cooling.

Comm/Data Building Block No. 2 (Figure 101) consists of equipment which permits VHF RF voice communications with a supporting CSM. The VHF link can also be used to communicate by voice with the ground in low-altitude, earth-orbit missions, the supporting CSM, and EVA crew activities. Equipment required to perform these functions includes: audio center, VHF/AM transceiver, VHF multiplexer, and VHF omniantennas of Block I or II. The Block I and II equipment varies slightly and is not interchangeable.

Physical characteristics of the equipment involved are as follows:

Audio Center; 10 pounds, 5.8H x 10.3W x 7.0D (inches)

Power Consumption; 20 watts dc

VHF/AM Transceiver; 12 pounds, 6.0H x 4.7W x 12.0D (inches)

Power Consumption; 40 watts dc

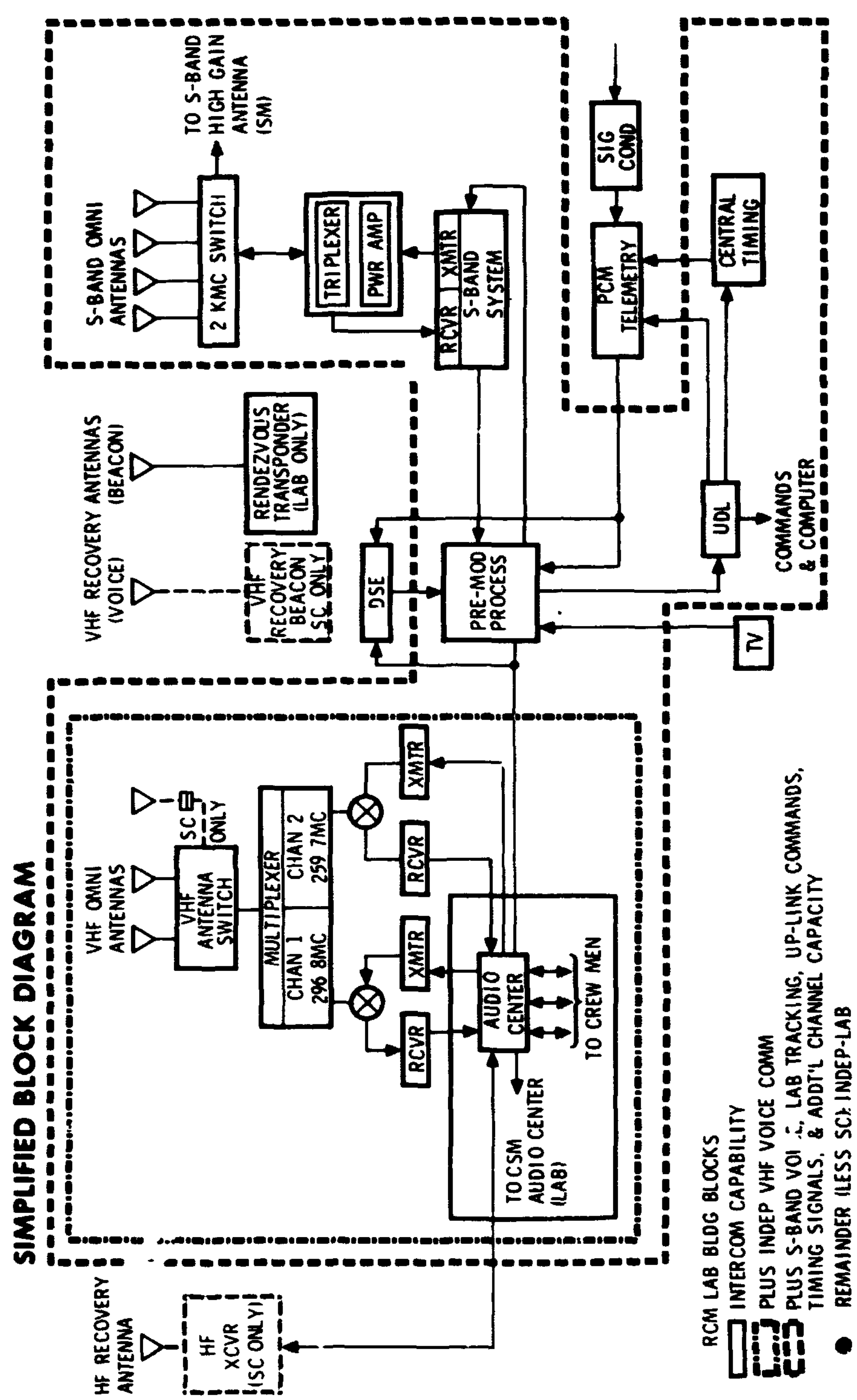


Figure 101. RCM Laboratory Communications/Data

VHF multiplexer; 2 pounds, 3.4H x 4.6W x 5.0D (inches)

No power required

Antennas (each) 12.5 pounds, 12.0H x 7.0W x 2.5 diameter (inches)

No power required

Installation constraints include the use of cooling plates for the audio center and the VHF/AM transceiver. Bl. .k II laboratory modules will require provisions for mounting the scim.tar antennas, since these antennas are mounted on the service module in typical Block II applications.

All equipment listed, except the antennas, is available from previous flights. The scimitar notch antennas are burnt beyond further use and the scimitar antennas are not recoverable.

Comm/Data Building Block No. 3 (Figure 101) provides S-band voice, coherent ranging, and up-data commands between the laboratory and earth. In addition, the hardline voice capability of Building Block No. 1 as well as timing signals to other laboratory subsystems, are provided. Data-channel capacity in excess of the above requirements is also available for wide band and subcarrier implementation. The above equipment is available from previous Apollo flights with minor refurbishments, except for SCIN antennas (Block I configuration) which are not refurbishable.

Physical characteristics of the Comm/Data Building Block No. 3 configuration is as follows:

·Unified S-band equipment (USBE); 38 pounds, 6.0H x 9.5W x 21.0D (inches)

Power consumption; 34 watts ac, 4 watts dc

S-band power amplifier; 30 pounds, 6.0H x 5.75W x 22.5D (inches)

Power consumption; 90 watts ac, 5 watts dc

Pre-Mod processor; 14.5 pounds, 6.0H x 4.7W x 10.6D (inches)

Audio center; 7.0 pounds, 6.0H x 4.7W x 8.7 diameter (inches)

Power consumption; 20 watts dc

Up-data link; 19 pounds, 5.6H x 8.9W x 17.75 diameter (inches)

Power consumption: 12 watts dc

Central timing equipment; 10 pounds, 6.0H x 4.7W x 8.0D (inches)

Power consumption; 20 watts dc

Antennas:

Block I SCIN (each, 2 required); weight-12.5 pounds, 12.0H x 7.0W x 2.5 diameter (inches)

Block II S-band omniantenna (each, 2 required); weight-2.0 pounds, 2.5" diameter x 4" long

No power required for antennas

Components of the Comm/Data Building Block No. 3 configuration will require cold-plate cooling for all items except the antennas. In addition, the S-band omniantennas, which are normally mounted in the aft heat shield, will require new mounting provisions if the heat shield is not used on laboratory structures.

Table 68 provides a summary of the equipment complement in each building block, the manufacturer, and the means of acquisition for utilization on the dependent laboratory modules.

DISPLAYS AND CONTROLS

Since displays and controls are a necessary function for operation of subsystems included in a laboratory vehicle, their number and type may be estimated easily from command module controls and displays. The location and placement of specific controls and displays is, however, a function of laboratory configurations, astronaut tasks, and human-factors consideration.

Three approaches may be considered in the RCM laboratory configuration study:

1. Use of existing panels in existing locations

2. Use of existing panels in new locations

3. Use of new panels in new locations

Approach number one is wasteful of space and ignores human factors but is a least-cost approach. All panels may remain in place with only those systems installed requiring operative controls and displays. All

Table 68. Communications/Data Building Blocks, Acquisition Identification

	Manufacturer	Means of Acquisition
Building Block No. 1		
1. Audio center	Collins Radio	Tested and reused from Block I or II S/C.
2. Hardline	NAA/SID	New item.
Building Block No. 2		
1. Audio center	Collins Radio	Tested and reused from Block I or II S/C.
2. VHF/AM transceiver	Collins Radio	Tested and reused from Block I or II S/C.
3a. VHF multiplexer (Block I)	Collins Radio	Tested and reused from Block II S/C.
3b. VHF triplexer (Block II)	Collins Radio	Tested and reused from Block II S/C.
4a. SCIN antennas (2, Block I)	NAA	New item (not refurbishable).
4b. SCIMITAR antennas (2, Block II)	NAA	New item (not recoverable).
Building Block No. 3		
1. Audio center	Collins Radio	Tested and reused from Block I or II S/C.
2. Up-data link	Motorola	Tested and reused from Block I or II S/C.
3. Pre-mod processor	Collins Radio	Tested and reused from Block I or II S/C.
4. Unified S-band equipment (USBE)	Collins Radio	Tested and reused from Block I or II S/C.
5. S-band power amplifier	Collins Radio	Replace TWT. Test and reuse from Block I or II S/C.
6a. SCIN antennas (2, Block I)	NAA/SID	New item (not refurbishable).
6b. S-band omniantenna (4, Block II)	AMECOM	Refurbished and reused from Block II S/C.
7. Central timing equipment	General Time	Tested and reused from Block I or II S/C.
8. Hardline	NAA/SID	New item.
Note: The Block I VHF/AM transceivers require new vacuum tubes prior to reuse.		

renovation data for RCM D&C will apply with the exception that only required items need be refurbished.

Approach number two will require rerouting of ships wiring and will add problems in layout but will provide added volume while maintaining minimum panel cost.

Tables 69 and 70 indicate those items and the refurbishment required for a fully independent laboratory on Blocks I and II. The Block I configuration has further eliminated four panels which retain either only one or a small number of switch functions which may be easily moved to another panel with blank spaces. Additional wiring is required at the new location.

Panel one of the Block II laboratory configuration may also be eliminated by the foregoing technique.

Approach number three will, most likely, be the eventual solution to the D&C layout since the space and volume requirements, plus the human factors requirements, will indicate that a new layout will be required for efficient operation in addition to new controls required by modified subsystems.

INSTRUMENTATION

Fully independent basic laboratory instrumentation will be identical with that of the RCM (see Section IX.).

Additional instrumentation may be supplied from a large list of qualified components which includes (but is not limited to):

Pressure Temperature Vibration Acceleration Displacement Strain Frequency Radiation Velocity Voltage Current Ph - Acidity	Sensors

Signal Conditioners
Power Supplies

Cameras (Film)
(TV)

Specific requirements for instrumentation will determine specific device to display, record, or telemeter information.

Table 69. Renovation Requirements for Block I (CM 012 and 014) RCM Laboratory Displays and Controls

Item	Test and Reuse	Not Required	Refurbish	Remarks
MDC Pan 1 V16-771401		X		
MDC Pan 2 V16-771402		X		Function to Panel 3
MDC Pan 3 V16-771403		X		
Pan 5 V16-771205		X		
MDC Pan 8 B16-771408			Replace 24 toggle switches Replace 2 connectors Replace wire harness	Requires engineering redesign (additional functions)
MDC Pan 10 V16-771410			Replace CW matrix Replace 2 connectors Replace wire harness	
MDC Pan 11 V16-77141.			Replace 6 toggle switches Replace 2 connectors Replace wire harness Replace CW matrix	Requires minor redesign
MDC Pan 12 V16-771212	X			
MDC Pan 13 V16-771413			Replace 25 toggle switches Replace 2 connectors Replace wire harness	
MDC Pan 15 V16-771415			Replace 14 toggle switches Replace 12 event indicators Replace 2 connectors Replace wire harness	
MDC Pan 16 V16-771416		X		Functions to Panel 8
MDC Pan 18 V16-771418			Replace 25 toggle switches Replace 2 connectors Replace wire harness Replace 15 event indicators Replace MA switch	
MDC Pan 19 V16-771419			Replace 2 toggle switches Replace 2 connectors Replace 1 event indicator Replace wire harness	
MDC Pan 20 V16-771420			Replace 26 toggle switches Replace 2 connectors Replace wire harness	
MDC Pan 21 V16-771221			Replace 7 toggle switches Replace 2 connectors Replace wire harness	
MDC Pan 22 V16-771422			Replace 10 toggle switches Replace 2 connectors Replace wire harness Replace 5 ckt breakers	
MDC Pan 23 V16-771423		X		Function to Panel 11
MDC Pan 24 V16-771424			Replace 1 toggle switch Replace 2 connectors Replace wire harness	
MDC Pan 25 V16-771425			Replace 1 toggle switch Replace 2 connectors Replace wire harness	
MDC Pan 26 V16-771426		X		Function to Panel 8
MDC Pan 100 V16-771498		X		
MDC Pan 200 V16-771497			Replace 2 toggle switches Replace 2 connectors Replace wire harness	

Table 70. Renovation Requirements for Block II RCM Laboratory Displays and Controls

Item	Test and Reuse	Not Required	Refurbish	Remarks
MDC Pan 1 V36-761011		X		Functions to Panel 2
MDC Pan 2 V36-761012			Replace 70 toggle switches Replace 19 event indicators Replace 2 C/W indicator matrices Replace 9 connectors Replace wire harness	Requires redesign (engineering) for added functions
MDC Pan 3 V36-761013			Replace 62 toggle switches Replace 21 event indicators Replace 1 MA switch Replace 10 connectors Replace wire harness	
MDC Pan 4 V36-761014		X		Functions to Panel 8
MDC Pan 7 V36-761017			Replace 3 toggle switches Replace 4 connectors Replace wire harness	
MDC Pan 5 V36-761015			Replace 3 toggle switches Replace 5 connectors Replace wire harness	
MDC Pan 8 V36-761018			Replace 10 toggle switches Replace wire harness Replace 5 connectors	Requires minor redesign
MDC Pan 6 V36-761016		X		
MDC Pan 9 V36-761019		X		
MDC Pan 10 V36-761020			Replace 5 toggle switches Replace 2 connectors Replace wire harness	
MDC Pan 11 V36-762065	X			
MDC Pan 13 V36-762013	X			
MDC Pan 98 V36-764098	X			
MDC Pan 99 V36-764099		X		
MDC Pan 97 V36-764090			Replace 8 toggle switches Replace 1 connector Replace wire harness	

NOTE: EL panel lighting may be removed entirely for laboratory configuration.

STABILIZATION AND CONTROL SYSTEM (SCS)

The following is a summary of the systems considered feasible for RCM Laboratory use. Many other systems such as TRW's OGO, which contains reaction wheels, GE's MAGS, which use gravity-gradient torques, and GE control-moment gyros were reviewed but not included as they could control only small spacecraft or were still under development. Of the following systems summarized below, only the Block II Apollo SCS and advanced inertial reference system are considered satisfactory for control of an RCM Laboratory. The Block II Apollo SCS, while requiring additional components for local vertical hold, offers the easiest installation. The Advanced Inertial Reference System, however, offers the longest mean-time between failure operation and lowest power draw.

Automatic Stabilization and Control System (ASCS).

Manufacturer

The system was manufactured by Minneapolis Honeywell, Aeronautical Controls Division and was used on the Mercury project.

Description of Functional Highlights

The ASCS provides automatic stabilization and orientation of the vehicle from the time of separation until the parachute is deployed. It consists of seven subassemblies: three rate gyros, two attitude gyros, an amplitude calibrator, and an acceleration switch.

The Mercury ASCS is considered unsatisfactory for RCM laboratory use because of the wide attitude deadband and the lack of manual control.

Functional Characteristics

The attitude deadband is ±1.0 degrees and the rate deadband is ±0.5 degrees. There is no provision for manual control.

Physical Characteristics

No information is available as to the physical characteristics.

Integration Constraints

The assemblies require cold plate cooling and the sensor assembly must be mounted so that the gyro axes are parallel to their respective axes.

Attitude Control and Maneuvering Electronics System

The system was manufactured by Minneapolis Honeywell, Aeronautical Division and was used on the Gemini project.

Description of Functional Highlights

The attitude control and maneuvering electronic system provides for stabilization and control of the spacecraft. The system converts rate and attitude signals from various sources to reaction control system signals.

The system consists of five devices: (1) attitude control electronics; (2) power inverter; (3) two gyro packages; and, (4) orbital attitude maneuvering electronics.

Attitude error inputs are obtained from three sources: (1) horizon scanner; (2) radar; and, (3) inertial platform and computer. Local vertical capability is, therefore, incorporated.

Functional Characteristics

The attitude deadband of the system is ±4.0 degrees. The system requires 92 watts at +28 VDC.

Physical Characteristics

The volume is 1000 cubic inches and is contained in five packages.

Integration Constraints

Cooling by cold-plate is required and the gyro assembly must be positioned so that the gyro axes are positioned parallel to their respective axes.

Lunar Module Stabilization and Control Subsystem (LM SCS)

The system was manufactured by Minneapolis Honeywell, Aeronautical Division as is used on the Apollo Lunar Module (Block I).

Description of Functional Highlights

The LM SCS provides the backup guidance to the primary guidance and navigation system (PGNS) and also provides the control electronics interface between the PGNS and the reaction control system and the propulsion subsystem.

The LM SCS consists of eight assemblies and a control panel:

1. Guidance coupler assembly
2. Rate gyro assembly
3. Attitude and translation control assembly
4. Descent engine control
5. Attitude controller
6. Translation Controller
7. Attitude controller
8. Backup guidance section

In utilizing the LM SCS Block I, consideration should be given to eliminating the descent engine control assembly and the translation controller.

Functional Characteristics

The system is designed with a goal of ±0. 1-degree attitude minimum dead band and 70 watts of power.

Physical Characteristics

There is no information presently available concerning physical char- characteristics

Integration Constraints

No information available.

Abort Guidance Section (AGS)

The AGS was manufactured by TRW Systems and is used on the Apollo lunar module (LM).

Description of Functional Highlights

The AGS is a backup guidance system used on the LM and is a strapdown guidance system which has the capability of maintaining attitude reference, computing attitude errors, driving attitude displays, solving guidance equations (digitally) and providing engine commands.

It consists of three assemblies: 1) abort electronics assembly (general-purpose digital computer), 2) abort sensor assembly (three strapdown gyros and three accelerometers), and 3) data entry and display assembly (display keyboard).

Utilization of the AGS in the RCM laboratory would require a reaction jet driver assembly.

Functional Characteristics

The three strapdown gyros provide output pulses proportional to incremental rotations of the vehicle about the body axis. The sensor scale factor is three arc seconds per pulse.

Physical Characteristics

The weight of the system is 61.6 pounds. It displaces a volume of 1797 cubic inches and is contained in three assemblies with a system operating power of 174 watts.

Integration Constraints

The sensor assembly must be mounted so that the gyro axes are positioned parallel to their respective axes.

Stabilization and Control System (SCS) Block I

The SCS is manufactured by Minneapolis Honeywell Aeronautical Controls Division and is used on the Apollo Block I Command Module.

Description of Functional Highlights

It provides stabilization and control of the Apollo Block I spacecraft and controls the RCS and SPS, based on rate and attitude inputs from the SCS sensors, manual inputs, or the G&N subsystem.

The SCS consists of the following assemblies:

1. Attitude gyro and accelerometer package (BMAG and accelerometer)

2. Rate gyro package (RGP)

3. Roll channel ECA

4. Pitch channel ECA

5. Yaw channel ECA
6. Attitude gyro coupler assembly (AGCU)
7. Display AGAP ECA (DECA)
8. Flight director attitude indicator (FDAI)
9. Attitude set/gimbal position display (AS/GPD)
10. Delta-V display panel
11. SCS control panel
12. 3-axis rotation control (2 each)
13. Translation control (2 each)

TVC capability is not a requirement of the RCM laboratory, therefore, the delta-V display panel and the two translation controls and one rotation control can be deleted from the subsystem.

The SCS will be capable of performing attitude hold, rate damping and manual maneuvering, utilizing the attitude reference subsystem. Local vertical capability is provided by an orbit rate generator (4.1 degree/ minute). The orbit rate is based on a 100-mile circular orbit. The lower equipment bay and the main display panel should be reconfigured to provide maximum utilization of the available space.

The utilization of the SCS Block I in the RCS laboratory is not recommended because of obsolescence, sparing capability and the fact that the units are not sealed and pressurized.

Functional Characteristics

The system has a maximum attitude deadband and a minimum attitude deadband capability. The maximum deadband is ±5.0 degrees and the minimum deadband is ±0.2 degrees. The rate deadband is ±0.2 degrees per second. The power requirements of the system are 147 watts at 400 Hz and 1.0 watts at +28 VDC (not including valve driver power).

Physical Characteristics

The SCS weight is 212 pounds, the LEB assemblies weigh 164 pounds, and the controls and displays weigh 48 pounds. Deletion of the delta-V display panel, two translation controls, and one rotation control will reduce

the SCS controls and displays weight to 33 pounds. The total system weight would then be 197 pounds.

The volume of the total system is 4970 cubic inches. The controls and displays volume will be reduced by 352 cubic inches with the deletion of the delta-V display panel, two translation controls, and one rotation control.

Integration Constraints

The following assemblies require cold-plate cooling:

1. Attitude gyro accelerometer package (AGAP)

2. Roll channel ECA

3. Pitch channel ECA

4. Yaw channel ECA

5. Display/AGAP ECA

6. Attitude gyro coupler assembly (AGCU)

7. Flight director attitude indicator (FDAI)

The attitude gyro accelerometer package and the rate gyro package must be mounted so that the gyro axes are positioned parallel to their respective axes

Stabilization and Control System (SCS, Block II)

The SCS is manufactured by Minneapolis Honeywell, Aeronautical Division and is used on the Block II Apollo command module.

Description of Functional Highlights

The SCS Block II provides stabilization and control of the Apollo Block II spacecraft and controls the RCS and SPS, based on the rate and attitude inputs from the SCS sensors, manual inputs, or the G&N subsystems.

The SCS consists of the following assemblies:

1. Solenoid driver assembly

2. Control electronics assembly

3. Display assembly

4. Gyro display coupler assembly

5. TVC servo amplifier assembly

6. Two gyro assemblies

7. Two flight director attitude indicators

8. Gimbal position and fuel pressure indicator

9. Attitude set control panel

10. Two rotational hand controllers

11. Translation hand controller

G&N and TVC capability is not a requirement of the RCM laboratory, therefore, the thrust vector position servo amplifier, gimbal position/fuel pressure indicator, translation control, one rotation control, and one FDAI can be deleted from the equipment list. The main display panel switches and circuit breakers associated with SCS control should be relocated to provide maximum utilization of space and operational capability. A number of the power and mode switches and circuit breakers must be provided with remote control in order to apply power and, subsequently, control moding for attitude hold and rate damping prior to docking to the unmanned laboratory.

If the vehicle is required to maintain local vertical, a means of local vertical attitude hold must be incorporated into the SCS. This can be accomplished with an orbit rate generator or a horizon sensor. The horizon sensor approach affords the best accuracy and flexibility.

Functional Characteristics

The system has a maximum attitude deadband and a minimum attitude capability. The maximum deadband is ±5.0 degrees and the minimum deadband is ±0.2 degrees. The rate deadband is ±0.2 degrees per second. The power requirements of the system are 240 watts at 400 Hz and 250 watts at +28 VDC. Gyros have a 2500-hour mean-time between failure.

Physical Characteristics

The SCS weight is 161 pounds, the LEB assemblies weigh 116 pounds, and the controls and displays weigh 45 pounds. Deletion of the thrust vector

position servo amplifier assembly from the equipment list will reduce the SCS LEB equipment weight to 102 pounds. Deletion of the gimbal position/ fuel position indicator, translation control, one rotation control, and one FDAI will reduce the weight of the controls and displays to 33 pounds. The total system weight would then be 135 pounds. The volume of the total system is: LEB 3496 cubic inches and controls and displays 1526 cubic inches. Deletion of the items discussed above will reduce the LEB volume to 948 cubic inches.

Integration Constraints

The following assemblies require cold-plate cooling:

1. Flight director attitude indicator (FDAI)

2. Solenoid driver assembly (SDA)

3. Control electronics assembly (CEA)

4. Display electronics assembly (DEA)

5. Gyro display coupler (GDC)

6. Thrust vector position servo amplifier assembly (SAA)

7. Gyro assembly No. 1

8. Gyro assembly No. 2

The gyro assemblies must be mounted so that the gyro axes are positioned parallel to their respective axes.

Advanced Inertial Reference System

The system is manufactured by the Autonetics Division of North American Aviation, Inc. It has not been utilized on any specific program to date.

Description of Functional Highlights

The system consists of an inertial reference assembly (IRA) utilizing an air-bearing gyro assembly (Autonetics G10B gyros) and a control electronics assembly. A horizon sensor can be incorporated for local vertical attitude hold. The mean time between failure of the gyro is 140, 000 hours. Manual control capability can be incorporated. As in other systems, a means of remote control of power and mode control will be required. A reaction jet driver assembly must be added to the existing configuration.

The inertial reference assembly is capable of maintaining a maximum drift of less than one arc degree per hour. The control electronics are presently mechanized for a high thrust of ±0.5 degrees per second and a low thrust of ±0.3 degrees per second.

The power requirements for the inertial reference assembly and the control electronics assembly are: Inertial Reference Assembly, 30 watts at 300 RPS gyro speed; 1 watt at 20 RPS gyro speed; control electronics assembly, 9.5 watts maximum; 5.0 watts average.

Physical Characteristics

The inertial reference assembly weighs 14 pounds and the controls electronics assembly weighs 3 pounds. The volume of the former is 539 cubic inches and the volume of the latter is 128 cubic inches.

Integration Constraints

With this system, the temperature must be maintained within a range of between 30 and 110 degrees fahrenheit.

The IRA must be mounted so that the gyro axes are positioned parallel to their respective axes.

REACTION CONTROL SYSTEM

The fully dependent RCM Laboratory, by definition, contains no reaction control system (RCS). It depends for attitude control and rotation on the CSM with which it is docked.

A fully independent laboratory, on the other hand, must have an RCS compatible with both the configuration and mission of the RCM. Since neither of these parameters is adequately known, no final system concept can be chosen. However, parametric data can be developed which can be used to optimize RCS design as the vehicle and its potential mission became better known. The following procedure has been used to develop this data:

1. A configuration has been assumed for a maximum weight RCM fully independent laboratory in order to calculate inertias for comparing the performance of candidate RCS concepts.

2. Basic equations to determine the propellant required to provide attitude hold and rotational capability to such a vehicle were developed.

3. Potential missions were assumed.

4. A series of candidate RCS concepts was developed and the parametric capabilities described.

5. The system weights necessary for candidate systems to perform assumed missions were calculated.

6. Implications of the calculations are discussed.

Assumed Configuration

The fully independent RCM Laboratory is assumed to have (Figure 102): (1) a forward end consisting of a truncated cone with diameters of 36 and 230 inches, an altitude of 120 inches, and a uniformly distributed mass of 8000 lb; (2) a cylindrical center section (RCM systems) 230 inches in diameter and 36 inches high with a uniformly distributed mass of 5000 lb; and (3) a cylindrical aft section (experiments) 230 inches in diameter and 60 inches high with a uniformly distributed mass of 12,000 lb.

The moments of inertia for such a laboratory are given in Table 71. Also given are moments for a total system consisting of this laboratory docked with a 23,134-pound (Spacecraft 103 type) CSM containing 3000 pounds of residual service propulsion system (SPS) propellant.

Basic Equations

Attitude Hold

When a vehicle is being maintained within a certain attitude deadband, an angular impulse (Lt) is applied whenever the deadband is approached to change attitude by $(\Delta\omega)$:

$$\Delta\omega = \frac{(Lt)\ \text{in.-lbf-sec}\ (32.17)\ \text{ft-lbm/sec}^2\text{-lbf}\ (180/\pi)\ \text{deg/radian}}{(I)\ \text{lbm-ft}^2\ (12\ \text{inch/ft.})\ (1/3600)\ \text{hour/sec.}} \qquad (1)$$

$$= 552{,}960\ \ Lt/I\ \text{deg/hour}$$

where

Lt = ηRFt, the number of thrusters firing (η) times their effective moment arm (R, inches) times their individual thrust (F, lbf) times the action time (t, sec)

I = moment of inertia (lbm-ft^2)

$\Delta\omega$ = change in angular velocity (deg/hour).

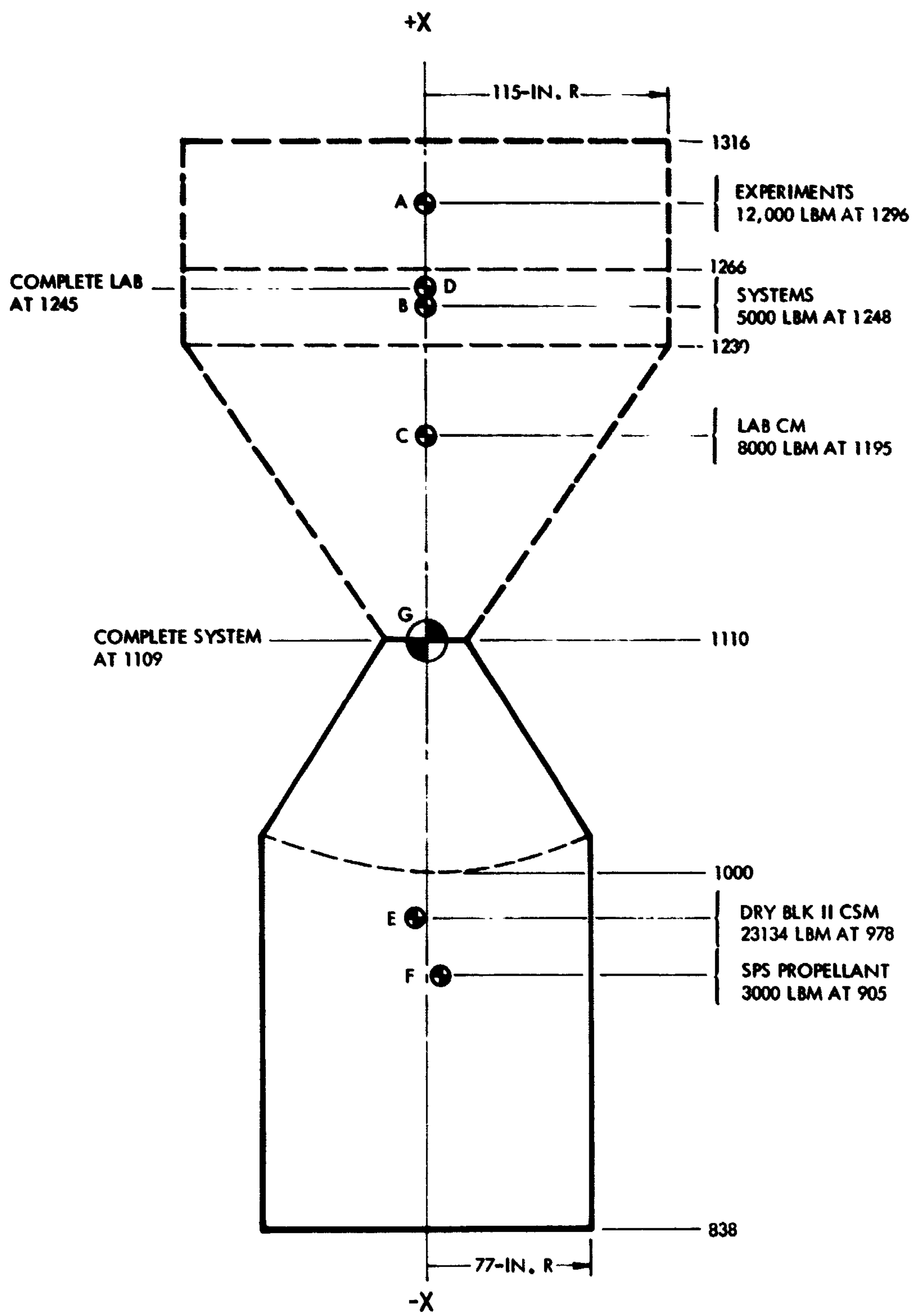


Figure 102. Inertial Mass Characteristics of the Fully Independent

Table 71. Fully Independent Laboratory Moments of Inertia

Code	Subsystem	Mass (lb)	Center of Gravity (in.) X	Y	Z	Inertia (Lbm $Ft^2 \times 10^{-6}$) I_x	I_y	I_z
A	Experiments	12,000	1296			0.561	0.3004	0.3004
B	Systems	5,000	1248			0.2295	0.1185	0.1185
C	Lab CM	8,000	1195			0.2175	0.1512	0.1512
D	Lab total	25,000	1254	0	0	1.008	0.912	0.912
E	CSM	23,134	978	-2.6	7.5	0.4227	1.2095	1.1863
F	SPS propellant	3,000	905	7.7	5.7	0.0488	0.0521	0.0593
G	System total	51,134	1110	-1.1	+3.7	1.488	9.363	[illegible].348

Sample Calculations:

$$I_{XA} = \frac{12000}{2} (9.503)^2 \times 10^{-6} = 0.516 \times 10^{-6} \text{ LBM FT}^2$$

$$I_{YA} = I_{ZA} = 12000 \left(\frac{9.583^2}{4} + \frac{5^2}{12} \right) = 12000\ (22.95 + 2.08) =$$

$$\overline{X}_D = \left| 1296\ (12000) + 1248\ (5000) + 1195\ (8000) \right| / 25000 = 1254$$

$$I_{YD} = I_{ZD} = 0.3004 + 0.1185 + 0.1602 + \frac{1}{144 \times 10^6} \left| 12000\ (1296 - 1254)^2 \right.$$

$$\left. + 5000\ (1248 - 1254)^2 + 8000\ (1195 - 1254)^2 \right|$$

$$- 0.5701 + 49.20/144 = 0.912$$

$$\overline{X}_G = \left| 1254\ (25000) + 978\ (23134) + 905\ (3000) \right| / 51134 = 1109$$

$$I_{YG} = 0.912 + 1.2095 + 0.0521 + \frac{1}{144 \times 10^6} \left| 25000\ (1254 - 1109)^2 \right.$$

$$\left. + 23134\ (1109 - 980)^2 + 3000\ (1109 - 905)^2 \right|$$

$$= 2.173 + 1035.4/144 = 9.363$$

The average rotational velocity will only be half this amount, or 276.480 Lt/I degrees per hour. If attitude is to be controlled to plus or minus degrees, the thrusters will be actuated every 2θ degrees of rotation, and the firing frequency (ff) will be:

$$ff = \frac{265,231 \text{ Lt/I}}{2\theta}$$

$$= 138,240 \text{ Lt/I} \quad \text{bursts per hour.} \tag{2}$$

Since the impulse per burst is (ηft lbf-sec), at a specific impulse (lbf-sec/lbm) the propellant consumption (Ca) is:

$$Ca = \frac{(nft)\,(138,240 \;\; nRft/I\theta)}{S}$$

$$= 138,240\ \eta^2 Rf^2 t^2/I\,\theta\ S \text{ lbm/hour.} \tag{3}$$

Since the propellant requirement is proportional to the square of the minimum impulse bit (nFt) applied by a system, propellant expenditure can be excessive if this minimum impulse bit is too great for the system inertia (I). To increase freedom in thruster selection it will be required that attitude hold be accomplished with single thrusters ($\eta = 1$). Since a configuration has been assumed (Table 71), propellant requirements for the independent laboratory (ca i) and the laboratory/CSM system (Cas), to maintain attitude about all three axes can be calculated:

$$Cai = \left[\frac{(0.13824)\,(10^6)\,(1)^2\,(Ft)^2\,(R)}{(10^6)\,(\theta\,S)}\right]\left[\frac{1}{1.008} + \frac{2}{0.912}\right] \tag{4}$$

$$= 0.4403\ (ft)^2\ (R)/\theta S \text{ lbm/hour}$$

$$Cas = \left[\frac{(0.13824)(10^6)(1)^2\,(ft)^2\,(r)}{(10^6)\,(\theta S)}\right]\left[\frac{1}{1.488} + \frac{1}{9.363} + \frac{1}{9.348}\right] \tag{5}$$

$$= 0.12245\ (ft)^2\ (R)/\theta S \text{ lbm/hour}$$

From the above it should be clear that a system designed to maintain attitude for the laboratory/CSM may be unsuitable to maintain attitude of the laboratory alone.

Rotation

The propellant required to initiate and conclude pitch, yaw, and roll rotation depends on the rotation rate desired, which in turn depends on the time one is willing to wait to complete a rotation. It shall be assumed that a rate of 0.3 degree per second (1080 degrees per hour) is satisfactory. This would permit 90 degrees of rotation in five minutes, assuming that thruster on-time is negligible compared to coast time. The thruster on-time required to initiate such a maneuver can be obtained by setting the ($\Delta\omega$) in Equation 1 equal to 1080 degrees per hour:

$$t = \frac{(1080)\ (I)}{(552,960)\ (nRF)} = 0.0019531\ I/(nRF) \qquad (6)$$

Since a complete rotation requires an equal impulse (nFt) to stop as well as start rotation, the propellant requirement (Cr) for a system of specific impulse (S) is:

$$Cr = 2n\ Ft/S = 0.0039062\ I/RS\ \text{lbm}. \qquad (7)$$

Since inertias have been assumed, it is easy to calculate the propellant required per set (one pitch, one yaw, one roll) of rotations for the independent laboratory (Cri) and the laboratory/CSM system (Crs):

$$Cri = (0.0039062)\ (2.832 \times 10^6)/RS = 11,062/RS\ \text{lbm/set} \qquad (8)$$

$$Crs = (0.0039062)\ (20.199 \times 10^6)/RS = 73,901/RS\ \text{lbm/set} \qquad (9)$$

It should be noted that the rotational propellant requirement is inversely proportional to moment radius, and the attitude hold requirement (Equation 3) directly proportional.

Minimum Thrust and Impulse Criteria

A minimum acceptable rotation rate has been assumed to be one that permits a 90-degree rotation in 5 minutes (ignoring acceleration/deceleration transients). Assume it is desired to spend not more than about 1/4 this time in acceleration and 1/4 in deceleration, with at least 1/2 the time used for the coast period between. If two thrusters are firing as a couple, the force from each thruster must be (from Equation 6):

$$F = \frac{0.0019531\ I}{RT} = \frac{(0.0019531)\ (9.363 \times 10^6)}{2\ (115\ \text{inches})\ (75\ \text{seconds})} = 1.06\ \text{lbf} \qquad (10)$$

Apparently a 1.0 lbf thruster is the minimum that can be used effectively with the laboratory CSM combination, even at a full 115-inch moment radius. If it is assumed that a 10-millisecond valve-open time is about the smallest desirable, 0.010 lbf-sec becomes the smallest minimum impulse bit for calculation.

Mission Assumption

Mission A: Laboratory Alone

The initial guideline in this study was that the RCS would only be used when the independent laboratory was separated from the CSM and unmanned, but that it might be required to support experiments during that period. Accordingly, it was assumed that the RCS would supply the following over a 30-day mission period:

1. Attitude hold in all three axes at ±0.2 degrees deadband (the limit of the Apollo SCS).

2. Two rotations about each axis daily.

The propellant requirement (C_A) for Mission A can then be obtained from Equations 4 and 8 as:

$$C_A = \left(\frac{0.4403\ (ft)^2\ (R)}{S}\ \frac{lbm}{deg\ hr}\right)\left(\frac{720\ hours}{0.2\ degree}\right) + \left(\frac{11062}{RS}\ \frac{lbm}{set}\right)(60\ sets) \qquad (11)$$

$$= (1/S)\ (1585.1\ (ft)^2 R + 663{,}720/R)\ lbm$$

Where

S = specific impulse (ft-lbf/lbm)

(ft) = minimum impulse bit (lbf-sec)

R = moment radius (inches)

Mission B: Laboratory and Laboratory-CSM

Midway in this study it was stated that the unmanned laboratory would not need attitude control to support experiments when it was separated from the CSM (although some rough control to prevent tumbling seemed desirable). The independent laboratory RCS, however, would be required to provide attitude control to the overall laboratory-CSM system for experimental

purposes while they were docked together. To simulate these conditions, the following mission (Mission B) is assumed:

1. Attitude hold of the laboratory alone in three axes at ±5.0 degrees deadband for 30 days.

2. Two rotations of the laboratory about each axis daily for 30 days.

3. Attitude hold of the laboratory-CSM system in three axes at ±0.2 degrees deadband for 30 days.

4. Two rotations of the total system about each axis daily for 30 days.

The propellant requirement (C_B) for Mission B can then be obtained from Equations 4, 5, 8 and 9 as:

$$C_B = \left[\frac{0.4403}{5.0} + \frac{0.12245}{0.2}\right] (ft)^2 (R/S) (720) + (11,062 + 78,901) (1/RS) (60)$$

$$= (1/s) (504.2 (ft)^2 R + 5,397,800/R) \text{ lbm.} \qquad (12)$$

Candidate Reaction Control Systems

Compressed ("Cold") Gas

The simplest reaction control system consists of a tank of compressed gas, pressure regulator, control valve(s), and simple supersonic nozzle(s). While low in specific impulse, such systems have proved reliable on space vehicles.

The specific impulse of a compressed gas acting through a converging-diverging nozzle may be estimated:

$$Isp = (\lambda C_D) C_F (RT/gM)^{1/2} \Phi(k) \qquad (13)$$

where

(λC_D), which corrects for nozzle half angle and discharge coefficient, is taken as 0.95

C_F, the thrust coefficient, is available in tables as a function of "k" and expansion ratio ϵ (the latter assumed to be 50).

T, the "chamber temperature" is assumed 70 F

R = 1545 ft-lbf/mol deg R

g = 32.17 ft-lbm/sec^2-lbf

M is molecular weight in lbm/mol

$k = C_p/C_v$

$$\Phi(k) = \left[(K) \left(\frac{2}{k+1}\right)^{\left(\frac{k+1}{k-1}\right)} \right]^{-1/2}$$

While this expression assumes a perfect gas and a constant value for "k," it is useful for comparative purposes. It was used (Table 72) to calculate the specific impulse available from 30 simple gases. A more important figure, however, is the impulse available from each pound of total gas-plus-container. This may be obtained from:

$$I_{sp}\,(sys) = I_{sp} / (1 + C/M) \tag{14}$$

Where C is an expression for tank mass (lbm per mol of gas) required to provide a 50 percent margin over ultimate strength (1.5 safety factor) at an assumed maximum temperature of 170 F. C equals 41.32 for oxygen and hydrogen (steel tank of 180,000 psi ultimate strength and 0.283 density) and 26.445 for all other gases (titanium tank of 160,000 psi ultimate strength and 0.161 density).

Application of this equation to the gases of Table 72 produces the following approximate system specific impulses:

1. About 43.0 for methyl ether, propane, and silane. All are flammable and somewhat toxic, and the first two liquify at temperatures (-13 and -48 F) to be expected in the expansion process.

2. About 37.2 and 36.3 for carbon dioxide and nitrogen, respectively, both nontoxic and nonflammable gases.

While silane (SiH_4) and carbon dioxide both deserve further analysis, nitrogen is chosen for further calculation because its use is well established in space systems. If it can be discharged using an effective moment radius of nine feet (108 inches) and a minimum impulse bit of 0.01 lb-sec, the nitrogen-plus-tank mass would be (from Equations 11 and 12):

Mission A: 170 lb

Mission B: 1377 lb

Essentially all of this mass is used for rotation, with attitude hold consuming less than a pound in each mission. If the moment radius can be increased by some factor (for example, by extendable reaction arms), the gas-plus-tank mass required would be reduced by the same factor.

"Tridyne" Catalyzed Gas Mixture

In the "Tridyne" concept being developed by Rocketdyne Division of NAA (References 19 and 20), propulsion is provided by a gas mixture of the following type:

Component	Mol Fraction	Mass Fraction
Nitrogen	0.85	0.9297
Oxygen	0.10	0.0625
Hydrogen	0.05	0.0078
Total	1.00	1.0000

The gas mixture has been shown to be non-explosive in these proportions. When the mixture is passed over MFSA catalyst (Engelhard Industries, Inc.), the hydrogen is oxidized and the mixture increases to about 1500 F, providing a specific impulse of about 150 lbf-sec/lbm when passed through a supersonic nozzle. Compositions providing temperatures and specific impulses much higher than this present problems not only because explosive limits are approached, but also because catalyst life begins to deteriorate rapidly at about 1700 F.

For short firing times (less than 0.2 second for a cold system) much of the released heat energy must be used to heat the catalyst. For attitude hold, then, the specific impulse should be calculated from the constituent mass fractions and the cold gas specific impulse values of Table 72.

Table 72

Gas	Molecular Weight (Lbm/Lbmol)	k=CP/CV @ 70°F	C_F @E=50	Isp (Lbf-Sec/Lbm)	Isp (sys) (Lbf-Sec/Lb
Hydrogen (H_2)	2.016	1.408	1.69	263.5	12.23
Helium (He)	4.003	1.66	1.55	157.2	20.7
Methane (CH_4)	16.042	1.316	1.74	98.5	37.2
Ammonia (NH_3)	17.032	1.317	1.74	95.4	37.3
Neon (Ne)	20,183	1.642	1.55	71.9	31.2
Acetylene (C_2H_2)	26.036	1.233	1.79	81.0	40.3
Nitrogen (N_2)	28.016	1.40	1.69	70.6	36.3
Ethylene (C_2H_4)	28.052	1.180	1.83	81.2	41.2
Ethane (C_2H_6)	30.068	1.220	1.81	76.5	40.5
Methylamine (CH_3NH_2)	31.058	1.185	1.83	77.0	41.6
Oxygen (O_2)	32.000	1.40	1.69	66.1	28.8
Silane (S_iH_4)	32.092	1.146	1.87	78.5	43.0
Argon (Ar)	39.944	1.67	1.55	51.0	30.7
Propane (C_3H_8)	44.094	1.130	1.89	67.9	42.4
Carbon Dioxide (CO_2)	44.01	1.304	1.74	59.5	37.2
Nitrous Oxide (N_2O)	44.016	1.311	1.74	59.5	37.2
Methyl Ether (C_2H_6O)	46.068	1.110	1.93	68.4	43.4
Methyl Chloride (CH_3Cl)	50.491	1.200	1.82	59.9	39.2
Cyanogen (C_2N_2)	52.036	1.256	1.76	56.0	37.2
Isobutane (C_4H_{10})	58.12	1.110	1.91	60.4	41.5
Ethyl Chloride (C_2H_5Cl)	64.517	1.190	1.83	53.4	37.8
Freon 23 (CHF_3)	70.018	1.191	1.83	51.3	37.2
Krypton (Kr)	83.7	1.689	1.54	34.9	26.6
Freon 22 ($CHClF_2$)	86.475	1.184	1.84	46.5	35.6
Freon 14 (CF_4)	88.01	1.159	1.86	47.0	36.1
Freon 13 ($CClF_3$)	104.47	1.145	1.87	43.4	34.6
Freon 12 (CCl_2F_2)	120.93	1.137	1.89	41.0	33.6
Xenon (Xe)	131.3	1.666	1.55	28.2	23.4
Freon 13B1 (CB_rF_3)	148.93	1.135	1.89	36.9	31.4
Freon 115 (C_2ClF_5)	154.48	1.091	1.92	37.3	31.8

*Sublimes at -109 F at 1.0 atmosphere, melts at -70 F at 5.2 atmo

FOLDOUT FRAME 2

Table 72. RCS Concepts and Their Characteristics

Isp (Lbf-Sec/Lbm)	Isp (sys) (Lbf-Sec/Lbm)	Isp (sys) Rank	Freezing Point °F	Boiling Point °F	Toxic ?	Fire Hazard ?
263.5	12.23	30	-434	-423	No	Yes
157.2	20.7	29	-458	-452	No	No
98.5	37.2	14	-296	-259	Yes	Yes
95.4	37.3	11	-128	- 28	Yes	Yes
71.9	31.2	24	-416	-410	No	No
81.0	40.3	8	-114	-119	Yes	Yes
70.6	36.3	17	-346	-320	No	No
81.2	41.2	6	-272	-155	Yes	Yes
76.5	40.5	7	-278	-127	Yes	Yes
77.0	41.6	4	-135	+ 19.8	Yes	Yes
66.1	28.8	26	-361	-297	No	No
78.5	43.0	2	-301	-170	Yes	Yes
51.0	30.7	25	-309	-302	No	No
67.9	42.4	3	-305	- 48	Yes	Yes
59.5	37.2	14	*	*	No	No
59.5	37.2	14	-152	-129	Yes	No
68.4	43.4	1	-216	- 12.8	Yes	Yes
59.9	39.2	9	-154	- 11	Yes	Yes
56.0	37.2	14	- 18.3	- 6.1	Yes	Yes
60.4	41.5	5	-230	+ 14.5	Yes	Yes
53.4	37.8	10	-218	+ 53.9	Yes	Yes
51.3	37.2	14	-247	-116	Yes	Yes
34.9	26.6	27	-251	-241	Yes	No
46.5	35.6	19	-256	- 41	Yes	Yes
47.0	36.1	18	-299	-198	Yes	Yes
43.4	34.6	20	-295	-115	Yes	Yes
41.0	33.6	21	-252	- 21.6	Yes	Yes
28.2	23.4	28	-169	-164	Yes	No
36.9	31.4	23	-270	- 73	Yes	Yes
37.3	31.8	22	-159	- 37.7	Yes	Yes

e, melts at -70 F at 5.2 atmospheres

$$I_{sp} = (0.9297)(70.6) + (0.0625)(77.0) + (0.0078)(263.5) = 72.5 \quad (15)$$

The gas must be stored in a steel tank, which would weigh about 1.613 lb per lb of gas. The "system specific impulse" of the tank-plus-gas therefore is about 57.41 as a hot gas and 27.74 as a cold gas. The "system mass" required for the reference missions can be computed from Equations 11 and 12 as

Mission A:

$$C_A = \frac{(1585.1)(0.01)^2(108)}{27.74} + \frac{663,720}{(108)(57.41)} = 108 \text{ lbm} \quad (16)$$

Mission B:

$$C_B = \frac{(504.2)(0.01)^2(108)}{27.74} + \frac{5,397,800}{(108)(57.41)} = 871 \text{ lbm} \quad (17)$$

Even when a modest adjustment for the mass of catalyst and its container (about 1.1 lb) is made, the Tridyne concept looks much better than nitrogen cold gas, and adds very little in complexity. However, it is not space-rated and requires considerable development effort.

Catalyzed Hydrazine

A recent S&ID report (Reference 21) states that anhydrous hydrazine, expanded through a 50:1 nozzle, has theoretical vacuum specific impulses varying between 213 and 248, depending on the extent of dissociation (between 100 and 40 percent, respectively,) of the product ammonia. Since hydrazine freezes at +34.5 F, its thermal control would present serious design difficulties to the RCM laboratory. A mixture of 31 percent water and 69 percent N_2H_4 by weight, however, freezes at -63 F; it is reported possible to design a catalyst system for use with such a mixture which would have only 20 percent NH_3 dissociation and would have a theoretical I_{sp} of about 190. This efficiency would not be realized in an actual system; 175 will be used arbitrarily as the I_{sp} for firings of several seconds or more.

Greater ammonia dissociation would be expected for very short pulses. Assuming it would be 50 percent, then:

$$\begin{array}{l} 3\,N_2H_4 \rightarrow 4\,NH_3 + N_2 + 144,300 \text{ Btu} \\ \underline{2\,NH_3 \rightarrow N_2 + 3\,H_2 - 39,600 \text{ Btu}} \\ 3\,N_2H_4 \rightarrow 2\,NH_3 + 2\,N_2 + 3\,H_2 + 104,700 \text{ Btu} \\ (96 \text{ LBM})\ (34 \text{ LBM})\ (56 \text{ LBM})\ (6 \text{ LBM}) \end{array}$$

Such a mixture at 70 F expanding as a cold gas should have an I_{sp} (using the cold gas specific impulses of Table 72) of:

$$\frac{34}{96}(95.4) + \frac{56}{96}(70.6) + \frac{6}{96}(263.5) = 91.4 \text{ lbf sec/lbm}$$

Since the mixture is only 69 percent N_2H_4 in the worst case assumption, the H_20 does not contribute to specific impulse.

$$I_{sp} \text{ (mixture)} = (0.69)(91.4) = 63.0$$

The propellant mass required for the assumed mission can then be calculated using expressions similar to Equations 21 and 22. This only represents propellant, and the total mass which logically should be compared with the gas-plus-tank mass of a cold gas or Tridyne system is found as follows:

	Mission A	Mission B
Propellant	35.4 lb	285.7 lb
(1) CM RCS fuel tank (45 lb capacity)	7.2 lb	
(3) LEM fuel tanks (104 lb capacity each)		30.5 lb
Gemini RCS helium tank	2.2 lb	
(3) Apollo RCS helium tanks		16.5 lb
Helium	0.2	1.5
Fuel solenoid valve(s)	1.0	2.0
Catalyst and containers	1.1	1.1
Total mass	47.1	337.3

Hydrazine systems of this type have been extensively developed, although not man-rated. Rocket Research Corporation has developed a 1.0 lbf engine (RRC-TA1 5-001) that might be used in the system, but further qualification and perhaps flight testing would be required.

90 Percent Hydrogen Peroxide

Reaction control was provided to the Mercury capsule by decomposition of 90 percent hydrogen peroxide. The Bell Model 8050 engine used in Mercury is reported to achieve a steady state specific impulse of 157 at 1.0 lb thrust, and to have an I_{sp} of about 80 at its standard pulse width of 0.15 second when the catalyst was cold (i.e., cold minimum impulse bit about 0.0764). Propellant requirements can then be calculated from Equations 6 and 7:

Mission A:

$$C_A = \frac{(1585.1)(0.0764)^2(108)}{80} + \frac{663720}{(157)(108)} \tag{18}$$

$$= 12.5 + 39.1 = 51.6 \text{ lbm}$$

Mission B:

$$C_B = \frac{(504.2)(0.0764)^2(108)}{80} + \frac{5397800}{(157)(108)} \tag{19}$$

$$= 4.0 + 318.3 = 322.3 \text{ lbm.}$$

System mass for comparison purposes would then be:

	Mission A	Mission B
Hydrogen peroxide	51.6 lb	322.3 lb
Mercury "Manual" toroidal tanks (2) at 30 lb capacity each	19.8 lb	
(11) at 30 lb capacity each		97.7 lb
Apollo RCS helium tanks at 5.5 lb each	5.5 lb	16.5 lb
Helium	0.5	1.5
Valves	2.0	4.0
Catalyst and containers	1.1	1.1
Total Mass	80.5	443.1

Bipropellant Systems

Storable bipropellant systems such as those used for reaction control on the Apollo command and service modules, LEM, and Gemini, provide a natural design alternative for the fully independent RCM Laboratory. A prime criterion for the suitability of such systems is the steady state and pulsed performance available from existing qualified engines. Table 73 itemizes the more important reaction engines of suitable size (5 to about 200 lb thrust) that have been used with storeable bipropellants in significant man-rated or space-rated systems:

Table 73. Available Bipropellant Engines

Application	Type Engine	Thrust Level (vacuum)	Manufacturer
Apollo CM RCS	Ablative	88.3 lb min.	Rocketdyne
Apollo SM RCS and LEM RCS	Radiation cooled	100 lb	Marquardt
Gemini RCS	Ablative	23.0 lb	Rocketdyne
Gemini OAMS	Ablative	23.0 lb, 79.0 lb 94.5 lb	Rocketdyne
Agena and Agena-GTV Secondary Propulsion Systems	Radiation cooled	16 lb, 200 lb	Bell
Surveyor	Regeneratively cooled	Throttleable (30-104 lb)	Thiokol RMD
Advent	Radiation cooled	25 lb	Marquardt
Titan III Transtage Attitude Control	Ablative	25 lb 45 lb	Rocketdyne

Ablative engines are desirable for reentry conditions where radiation cooling is not feasible, and are best suited to shorter total firing times that do not exceed the cooling capability of the ablative material. The suitability of the ablative engines to the RCM Laboratory application can only be assessed when more is known of potential mission requirements. Radiation cooled engines are lighter and should be suitable for the environmental requirements of the most probable RCM Laboratory missions.

The Marquardt 100-lb thrust engine used on the Apollo service module and LM RCS has a steady state specific impulse of about 270, and a specific impulse of about 135 at a minimum impulse bit of about 0.5 lbf-sec when operated by the Apollo stabilization and control system (SCS). Equations 11 and 12 can be used to obtain the propellant requirement:

Mission A:

$$C_A = \frac{1585.1\ (0.5)^2 R}{135} + \frac{663720}{270R} \tag{20}$$

$$= 2.935R + 2458/R$$

$$= 317.0 + 22.8 = 339.8 \text{ lbm. at } R = 108 \text{ in.}$$

Mission B:

$$C_B = \frac{(504.2)\ (0.5)^2 R}{135} + \frac{5397800}{270R} \tag{21}$$

$$= 0.9337R + 19992/R$$

$$= 100.8 + 185.1 = 285.9 \text{ lbm at } R = 108 \text{ in.}$$

Clearly the high thrust level and minimum impulse bit of the SM RCS engine results in undesirably high propellant consumption for attitude hold (and the rotational velocities in attitude hold are greater than may be desired for some experiments). One way to reduce propellant requirements is to reduce the effective moment arm. (This slows the rotation rate and therefore reduces the frequency at which the engines must fire.) If Equation 20 is differentiated with respect to R and set equal to zero,

$$\frac{dC_A}{d_R} = 0 = 2.935 - 2458/R^2 \tag{22}$$

$$R = (2458/2.935)^{1/2} = 28.94$$

When the moment arm is decreased from 108 inches to 28.94 inches, e propellant requirement for Mission A is reduced from 339.8 to 169.9 lb, equally divided in use between attitude hold and rotation. This reduction may be achieved either by placing the engines closer to the center of gravity or by turning them so that only a small sine function of thrust is effective. The same change would increase the propellant required for Mission B from 285.9 to 717.8 lb.

A better solution seems to be the use of an engine of smaller thrust. Two radiation-cooled engines are available: the 16-lb engine used in the Agena-GTV secondary propulsion system, and the 25-lb engine used in the Advent program. If the minimum impulse bit can be assumed proportional to engine thrust, the propellant requirement using the Agena engine should be:

Mission A:

$$C_A = \frac{(1585.1)\,(0.08)^2\,(108)}{135} + \frac{663720}{(260)(108)} = 8.1 + 23.7 = 31.8 \text{ lbm} \qquad (23)$$

Mission B:

$$C_B = \frac{(504.2)\,(0.08)^2\,(108)}{135} = \frac{5397800}{(260)(108)} = 2.6 + 192.2 = 194.8 \text{ lbm} \qquad (24)$$

"System masses" for comparison with cold gas and monopropellant systems are as follows:

Engine	Component Mass, LB				
	SM RCS			Agena	
Mission	A	A	B	A	B
Moment arm	108"	28.94"	108"	108"	108"
Oxidizer	194.4	104.5	180.4	20.3	129.6
Oxidizer tankage	11.5	9.2	11.5	3.5	9.2
Fuel	145.4	65.3	105.5	11.4	65.2
Fuel tankage	18.3	8.3	11.5	3.5	8.3
Helium tankage	11.0	5.5	11.0	2.2	5.5
Helium	1.0	0.5	1.0	0.2	0.5
Propellant valves	3.0	2.0	2.0	2.0	2.0
System mass	384.6	195.3	322.9	43.1	220.3

Bipropellant Components

Although certain assumptions on propellant tank sizes have been made to arrive at the foregoing mass comparisons, component selection and system design cannot really begin until the mission requirements are better understood. However, a range of qualified, man-rated components is available from the Apollo, LM, Gemini, and Agena programs. Propellant tanks of similar design (titanium shell and teflon bladder) qualified for bipropellants are available in a wide range of sizes (440 to 3900 cubic inch capacity). For missions longer than 30 days, however, the adequacy of this tank design has not been demonstrated. Some alternative tank design such as the bellows concept used on the Agena-GTV secondary propulsion system may be necessary for longer missions.

Steel and titanium spherical pressure vessels for compressed gases are available in a wide range of sizes. Selection of valves, pressure regulators, and similar components would be made, where possible, from existing space-rated systems. Such selection, however, requires a knowledge of the mission and the expected thermal environment and a careful analysis of the extent of qualification of available components which is beyond the scope of the present effort.

Summary of Analysis

System Mass Comparisons

The "system mass" required to perform the two assumed missions with the several RCS concepts discussed above are summarized in Table 74.

Table 74. System Mass Comparisons

RCS Concept	System Mass Required	
	Mission A	Mission B
Compressed nitrogen	170	1377
Tridyne (catalyzed N_2-O_2-H_2 mixture)	109	872
Catalyzed hydrazine (69/31 mixture with water)	47	337
Hydrogen peroxide (902)	80	433
Bipropellant, using 100-lb SM RCS engine	385*	323
Bipropellant, using 16-lb Agena engine	43	220

*195 lbm when moment radius is reduced to 29 inches.

The "system mass" totaled includes the propellant, pressurizing gas, catalyst, and containers for them, plus control valves for the liquid propellants. The totals do not include gas regulation components (pressure regulator, check valves, relief valves, etc.), fill/vent provisions, and fluid tubing, all of which should be somewhat comparable for all systems. The totals also do not include thruster mass, which will vary from the 3- to 5-lbm per liquid engine down to a much more modest mass for the simpler 1.0-lbf gas or monopropellant thrusters.

Discussion

Compressed gas systems are attractive because of their extreme simplicity and because they avoid the problem of propellant freezing. Where only attitude control of a fairly stable system (even a large one) is required, they represent a good solution. However, when many rotational maneuvers of a system with substantial inertia is required, the low specific impulse of compressed gases soon causes mass requirements to become excessive. The Tridyne concept offers the same advantages at about 2/3 the mass requirement, but the concept is still in the developmental stage.

The catalyzed hydrazine system is attractive from a mass requirement standpoint, and the concept has been extensively developed. Some additional testing would be required, especially if use of a hydrazine-water mixture was chosen to minimize the freezing problem. The hydrogen peroxide system was man-rated on Project Mercury, but the components are out of production and some replacement probably would be necessary for the toroidal propellant tanks, which would be hard to locate and insulate. These factors, together with the lower performance of the peroxide system, make it seem a less desirable choice than catalyzed hydrazine.

A storeable bipropellant system offers the best specific impulse of the choices considered, and therefore potentially the lowest system mass. However, a system employing the large (100-lbf) engines used on the Apollo SM and LM RCS appears too powerful for efficient long-term narrow-deadband attitude control. When the smaller 16-lbf Agena engine is substituted, however, mass requirements are reduced dramatically.

Conclusion

It appears from this analysis that a bipropellant system employing an engine of modest thrust (such as the 16-lb engine used in the Agena GTV) is the most efficient, lightest concept for the missions assumed. While use of the Apollo SM RCS feed system with this engine has been assumed for preliminary pricing purposes, use instead of the Agena components and system should be considered in a further phase. In particular, use of the Agena bellows-type propellant tanks may be desirable as mission durations exceed 30 days.

No final recommendation can be made, however, until mission requirements are better understood. At that time the desirability of the bipropellant system should be reassessed and compared with alternatives such as the catalyzed hydrazine-water monopropellant concept.

RCS Building Blocks

Service Module RCS

Descriptive Title. Apollo Service Module Reaction Control Subsystem, Block I.

Manufacturers and Availability Status. The SM RCS subsystem is assembled, installed, and checked out by NAA/S&ID. Components are obtained from the suppliers indicated in Table 75. The qualified components used on the final Block I subsystem are essentially the same as those used on the Block II subsystem, and should be available at the time the RCM laboratory is implemented. Tooling and capability to assemble the subsystem should still be available at S&ID, even though the Block I subsystem no longer will be in actual production. It is further assumed that components can be procured on a non-interference basis with Apollo requirements, but no investigation of schedule impact has been made.

Space Programs. The configuration discussed herein is that used on the final Block I vehicles of the Apollo program.

Description of Functional Highlights. The function of the SM RCS is to provide the thrust required for three-axis stabilization and control of the independent RCM Laboratory throughout its mission. In addition to the stabilization and control function, the SM RCS is capable of providing translational velocity increments, minor velocity corrections, and emergency retrograde from earth orbit.

The SM RCS is pulse-modulated, pressure-fed by helium gas, and utilizes storeable hypergolic liquid propellants (N_2O_4 and MMH). Normally the SM RCS is controlled automatically by the guidance and control system, but override controls are provided for manual operation.

Functional Characteristics. The SM RCS engines will provide reaction thrust to the Apollo CSM using 16 engines. Each engine is a radiation-cooled, pulse-modulated, pressure-fed, bipropellant thrust generator designed to produce a vacuum thrust of 100 pounds (Figure 103). Each engine consists of two propellant control valves, an injector, and a thrust chamber. The valves are solenoid-operated and control the flow of the respective propellants to the engine. The combustion chamber is made of unalloyed molybdenum and is coated inside and out with molybdenum disilicide to prevent oxidization. A thrust chamber nozzle is attached to the combustion chamber at a nozzle expansion ratio of 7:1; the expansion ratio of the nozzle exit is 40:1. Performance values are shown in Figures 104 and 105. Other RCS components are listed in Table 76 .

Table 75 . Components of Apollo SM RCS

Component	Supplier
SM RCS engine	Marquardt Corporation
SM quad heater	Thermal System
Propellant solenoid valve	National Water Lift
Relief valve	Calmec Manufacturing Company
Check valves	APCO
Helium pressure regulator	Fairchild-Hiller Corporation
Helium solenoid valve	National Water Lift
Fuel tanks	Bell Aerosystems
Oxidizer tanks	Bell Aerosystems
Helium pressure vessel	Menasco Company
Dynatube fittings	Resistoflex Corporation
Propellant fill and drain fitting	J. C. Carter Company
Helium coupling	Purolator Products
Test Point coupling	Lear Seigler
Propellant filter	Cemarc Corporation

Physical Characteristics. The SM RCS consists of four independent modular quad assemblies located at 90 degree increments around the periphery of the SM. Each modular assembly contains a helium storage and distribution system, an oxidizer storage and distribution system, a fuel storage and distribution system, and an engine assembly cluster with four engines. A schematic of an individual quad is shown in Figure 106 and a perspective drawing in Figure 107 Each quadrant is currently mounted on a 36 x 67 inch door as shown in Figure 107.

Integration Constraints. Present Apollo requirements require that temperatures be controlled to the values shown in Table 77.

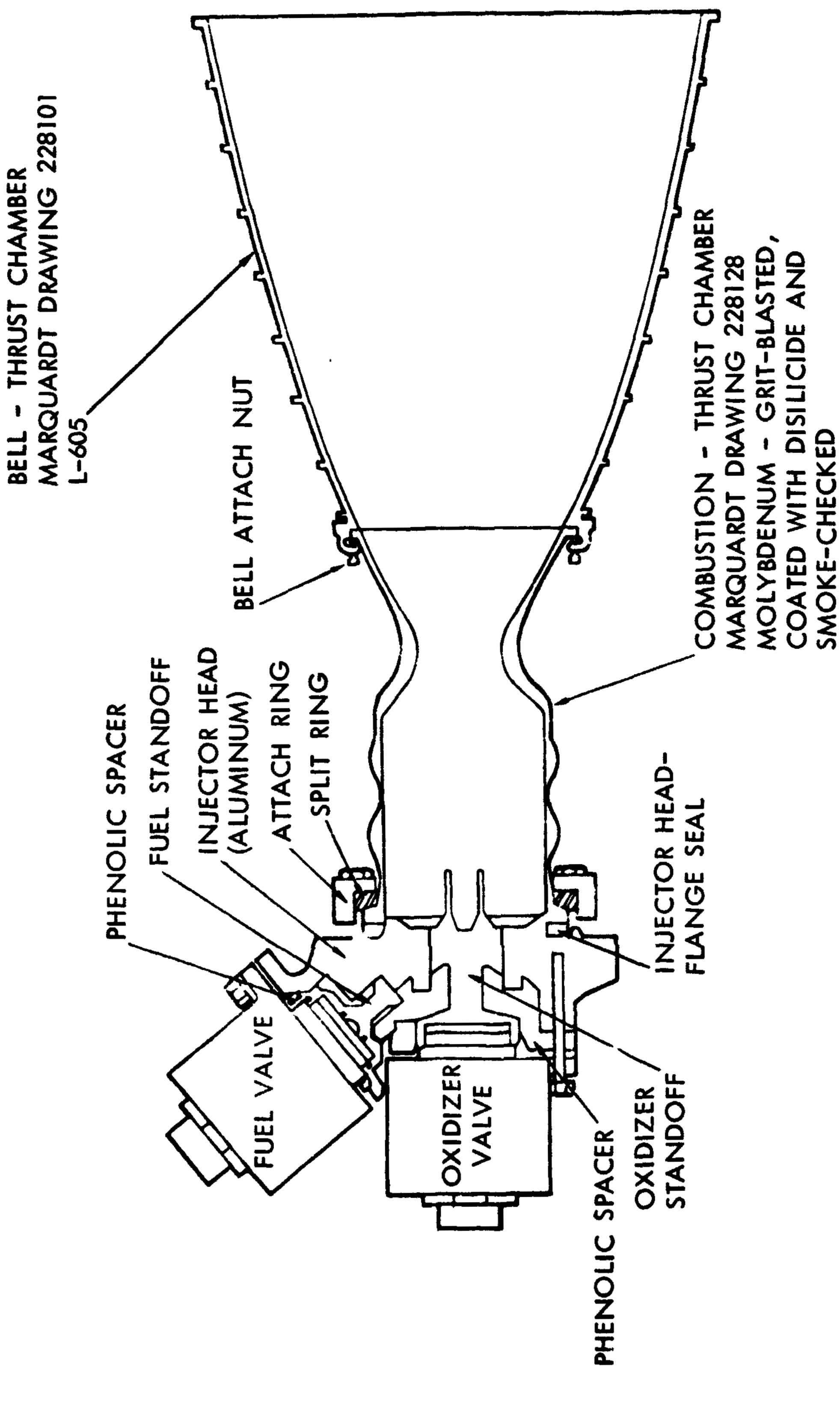


Figure 103. Apollo SM RCS Engine

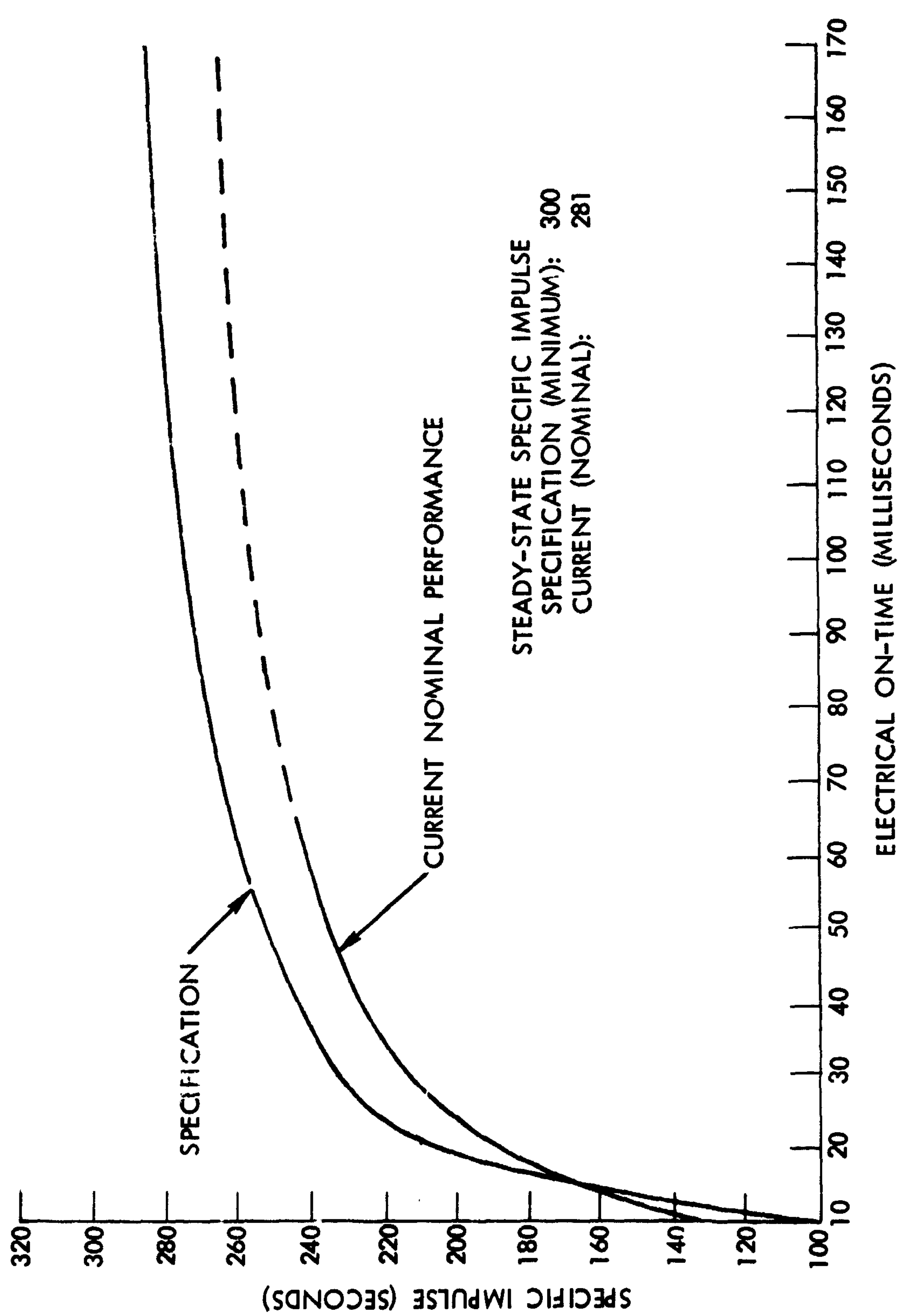


Figure 104. Specific Impulse Versus Pulse Time for SM RCS Engine

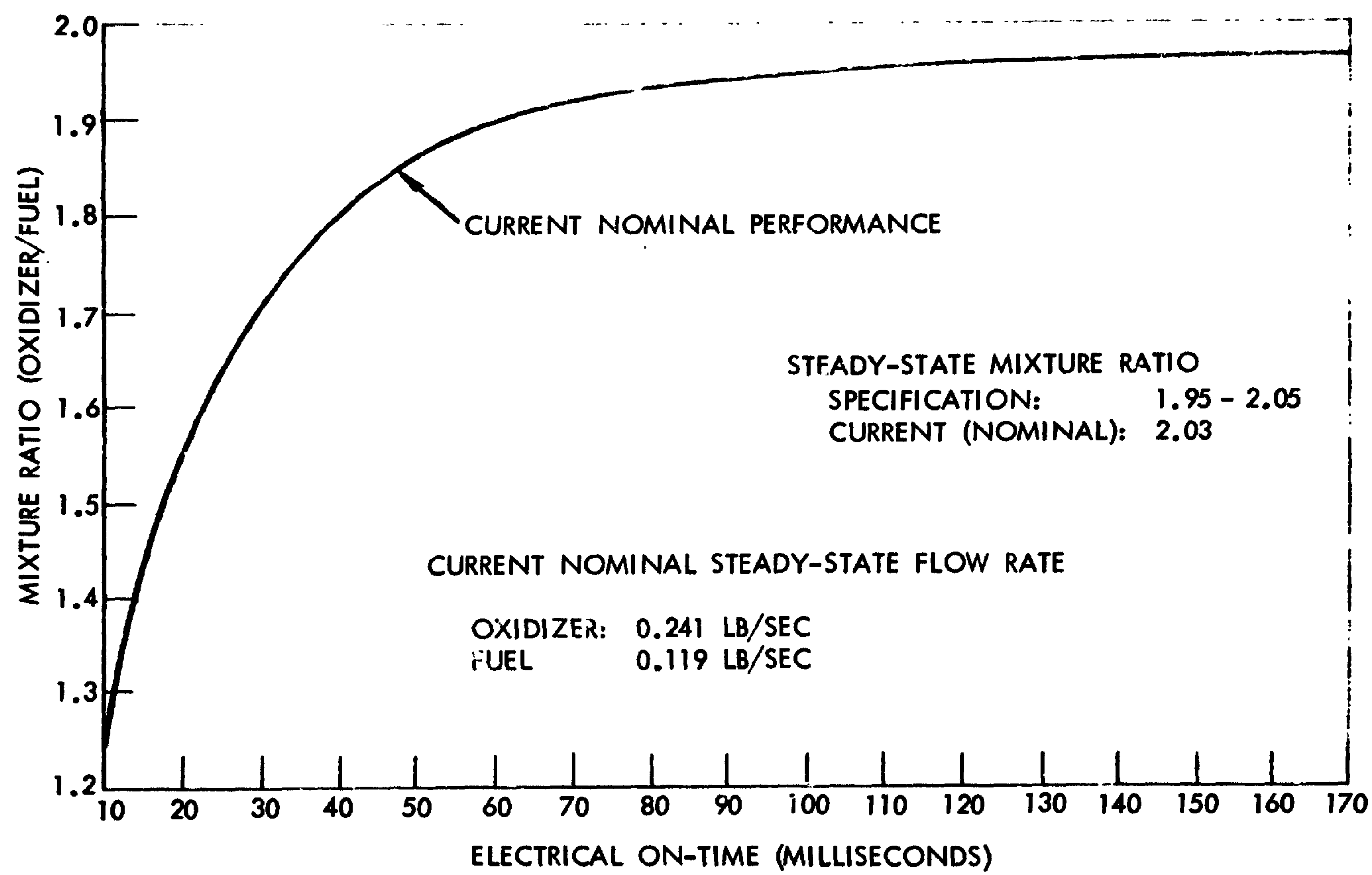


Figure 105. Mixture Ratio Versus Pulse Time for SM RCS Engine

Table 76 . Description of Apollo SM RCS Components

Component	Function	Type	Parameters
Fuel Tank	MMH Storage	Titanium · positive expulsion	Capacity - 37.5 Usable at 65 F and 30 PSIG
Oxidizer tank	N_2O_2	Titanium - positive expulsion	Capacity - 75 lb Usable at 65 F and 30 PSIG
Helium vessel	He Storage	Titanium	Proof press 6667 PSIG Burst press 7500 PSIG
He isolation valve	Shut off or malfunction isolation	N/O, latching, solenoid	
Pressure regulator	Reduce and regulate Helium pressure	Series/parallel	Tank press 5000 PSIG Regulate to 181 PSIG
Check valve	Prevent propellant from moving upstream	Differential pressure actuated	
Pressure relief valve	Prevent over-pressurization of system.	Spring loaded Relief with burst disc	Burst 340 PSIG Relief 346 PSIG Reseat 327 PSIG
Propellant valve	Isolate malfunctioning engine or shut-off	Solenoid	
Propellant vilter	Prevent large particles from entering engine	Propellant compatible filter material	
Disconnect couplings	Test points for ground operations	Quick-disconnect, poppet with flight cover	
He fill coupling	Fill and drain He during ground operations	Quick-disconnect, poppet with flight cover	
Propellant fill coupling	Fill, drain and vent during ground operations	Quick-disconnect, poppet with flight cover	
Quad heater	Prevent propellant freezing	Electrical resistance	RCM Lab application may require larger heater

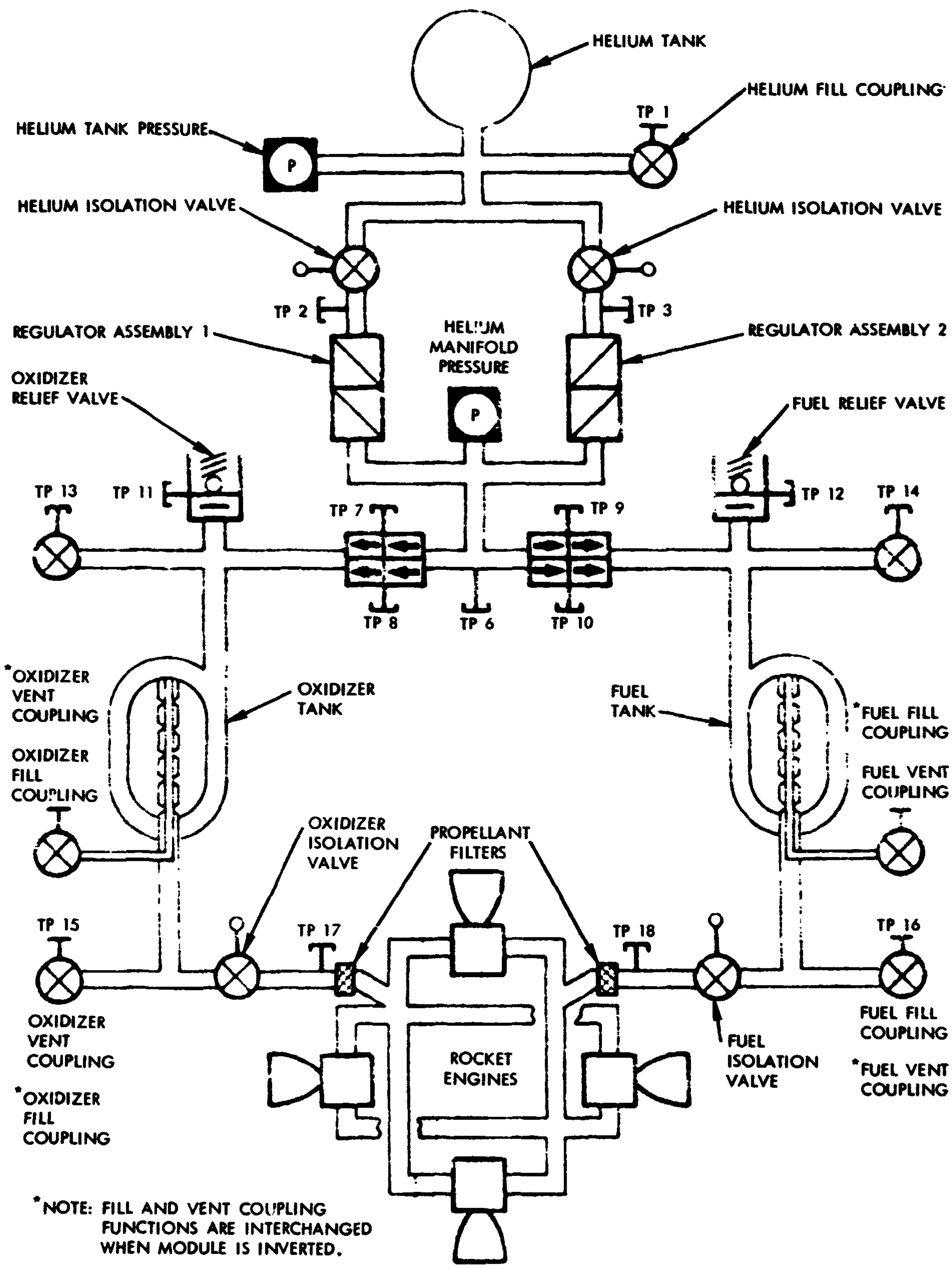


Figure 106. Schematic of Typical SM RCS Block I Quad

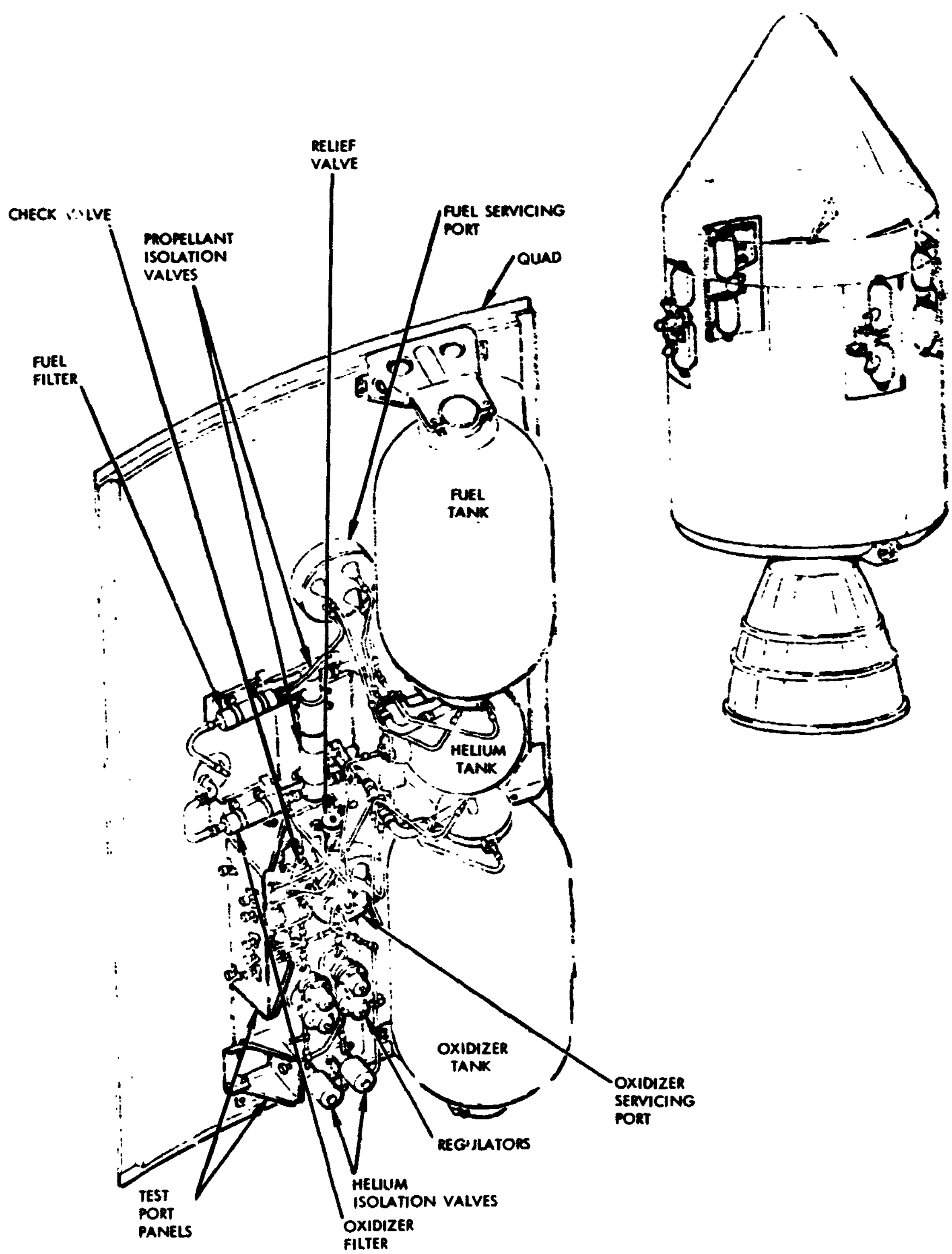


Figure 107. Perspective of Typical SM RCS Block I Quad

Table 77. Critical CSM RCS Temperature Limits

Components	Temperature Limits, °F		Remarks
	Maximum	Minimum	
SM RCS			
Engine fuel valve	175 (1)	35 (2)	
Engine oxidizer valve	175 (1)	35 (2)	
Injector head	350 (3)	32 (4)	
Fuel tank	85 (5)	40 (4)	
Oxidizer tank	85 (5)	40 (4)	
Helium check valve	150 (6)	30 (7)	
Fuel isolation valve	105 (8)	40 (4)	
Helium in storage tank	150 (6)	40 (7)	
CM RCS			
Engine fuel valve	200 (1)	40 (2)	1-second firing permitted at 20 F
Engine oxidizer valve	200 (1)	40 (2)	1-second firing permitted at 20 F
Fuel tank	105 (5) (8)	40 (4)	
Oxidizer tank	105 (5) (8)	40 (4)	
Fuel isolation valve	105 (8)	40 (4)	
Oxidizer isolation valve	105 (8)	40 (4)	

(1) To preclude two-phase propellant flow and/or seat damage.
(2) To preclude freezing of propellant and/or assure valve operation on direct coil.
(3) Limit for shrink fit of insert into injector body.
(4) To preclude freezing of propellant.
(5) To preclude overpressurizing system.
(6) To preclude overpressurizing helium system
(7) To preclude low feed pressure
(8) Material exposure limit.

Application to RCM Laboratory. The 100-lb engines of the SM RCS appear to be too powerful for efficient attitude control of the RCM laboratory. Substitution of smaller engines such as the 16-lb Agena GTV engines reduces propellant requirements substantially. Also, the capacity of the four SM RCS quads may be more than needed for any expected mission. Substitution of CM RCS for SM RCS propellant tanks reduces this by about 1/3, which may be more in keeping with requirements.

The SM RCS quads are designed to fit into the SM RCS structure, and the large heat sink provided by the large service propulsion system tanks helps maintain thermal control. If the quads are to be mounted on the ends of the RCM laboratory support structure, careful analysis of thermal control requirements will be necessary. Repackaging the system into a more compact geometry may be necessary to minimize heat loss.

Command Module RCS

Descriptive Title. Apollo Command Module Reaction Control Subsystem, Block II,

Manufacturers and Availability Status. The CM RCS subsystem is assembled, installed, and checked out by NAA/S&ID. Components are obtained from the suppliers indicated in Table 75. It is assumed that the qualified components and subsystems similar to those used on Block II will be available at the time the RCM laboratory is implemented. Tooling and capability of the suppliers to produce such components will still exist even though the components no longer may be in actual production. It is further assumed that the components can be procured on a non-interference basis with Apollo requirements, but no investigation of schedule impact has been made.

Space Program. The configuration discussed herein is that used on the Block II vehicles of the Apollo program.

Description of Functional Highlights. The function of the CM RCS is to provide the thrust and impulse required for three-axis rate damping of the RCM Laboratory during entry. The system is activated only for the earth entry phase. The CM RCS is pulse-modulated, pressure fed by helium gas, and utilizes storeable hypergolic liquid propellants (N_2O_4 and MMH). Normally, the CM RCS is controlled automatically by the guidance and control system but override controls are provided for manual operation.

Functional Characteristics. Two identical systems, designated A and B, are provided, which normally operate simultaneously; however, if one system fails, the remaining system is capable of adequate control for entry. Each assembly consists of a pressurization subsystem, a propellant subsystem, a

dump and purge subsystem, and six rocket engines with nozzle extensions. The CM RCS engines will provide reaction thrust for attitude control of the RCM Laboratory during entry. Each engine is an ablative-cooled, pulse-modulated, pressure-fed, bipropellant thrust generator designed for a minimum vacuum thrust of 88.3 pounds at an expansion rate of 9:1 (without nozzle extension). The nozzle extension provides a passage for the exhaust gases through the heat shield. The two basic components of the engine are the propellant control valve assembly and the thrust chamber assembly, which includes the injector, combustion chamber, nozzle, and mount. Performance values are shown in Figures 108 and 109. Other CM RCS components are listed in Table 78.

Physical Characteristics. The complete CM RCS, shown schematically in Figure 110 and perspectively in Figure 111, consists of two equally capable and identical subsystems (designated A and B) located outside the CM pressure hull in the aft compartment. These assemblies are supported by displays and controls inside the CM pressure hull.

Integration Constraints. Present Apollo requirements require that temperatures be controlled to the values shown in Table 77.

Application to RCM Laboratory. In an RCM laboratory application the dump and purge provisions of the CM RCS would be unnecessary and should be deleted. The ablative CM RCS engines do not seem appropriate to the RCM laboratory application and would best be replaced by smaller radiation-cooled engines. Since the CM RCS consists of two identical and independent assemblies, one assembly could be packaged as a "quad" at each end of the RCM support cruciform. This would provide 2/3 as much propellant as four SM RCS quads, which is probably more than adequate. Alternatively, the CM RCS could be used at its present capacity (1/6 the SM RCS capacity in each of two assemblies), with each independent assembly feeding two clusters of three or four engines each at the ends of the cruciform structure.

The CM RCS is now distributed throughout the aft equipment bay of the Apollo command module in order to fit within the geometric limitations of the heat shield, and is protected from temperature, radiation, and micrometeorites by the thick heat shield. If the protection of this heat shield were removed (as in the RCM Laboratory), the subsystem would have to be repackaged. Particular attention would have to be paid to the thermal control of long, small diameter propellant lines.

Agena GTV SPS

Descriptive Title. Model 8250 Secondary Propulsion System for the Agena Gemini Test Vehicle.

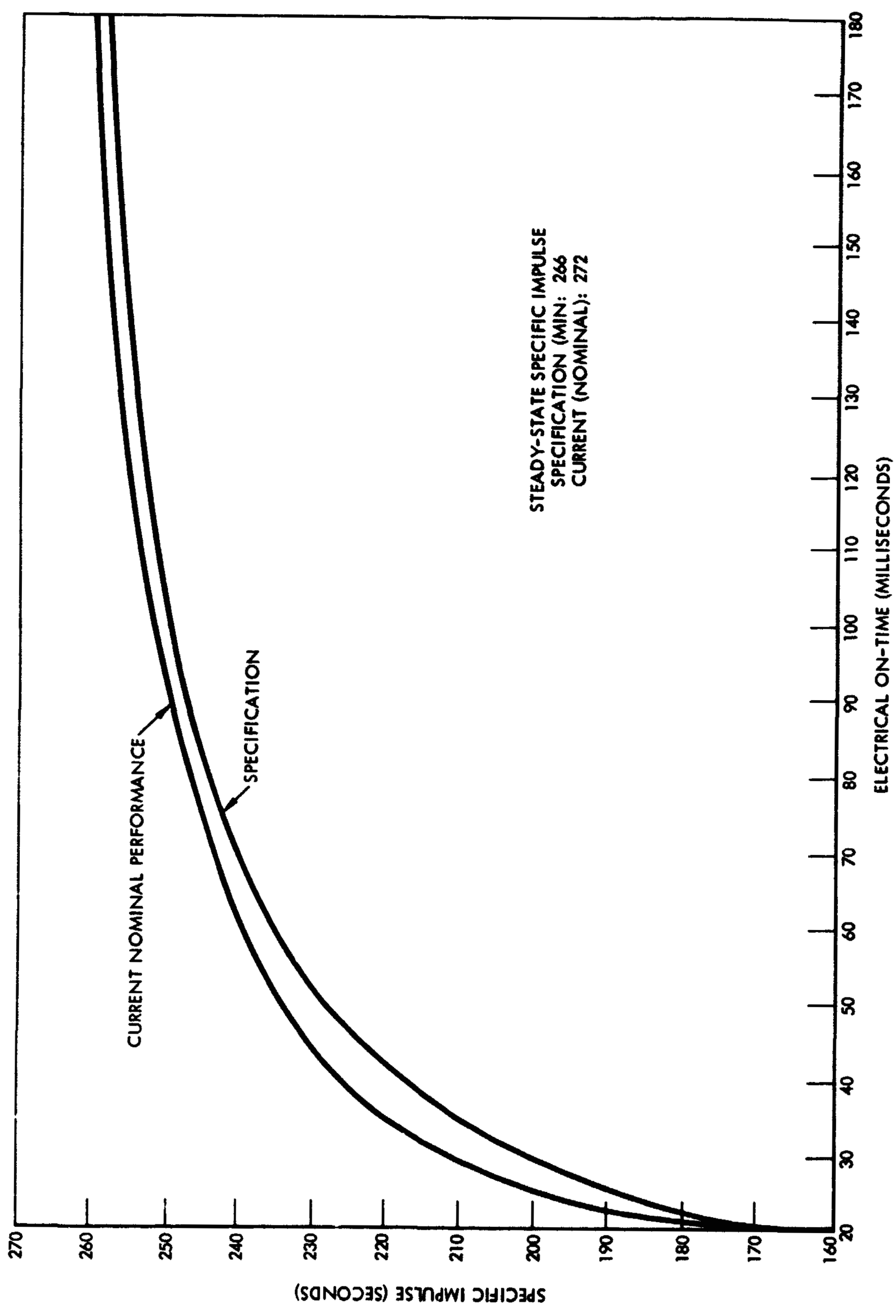


Figure 108. Specific Impulse Versus Pulse Time for CM RCS Engine

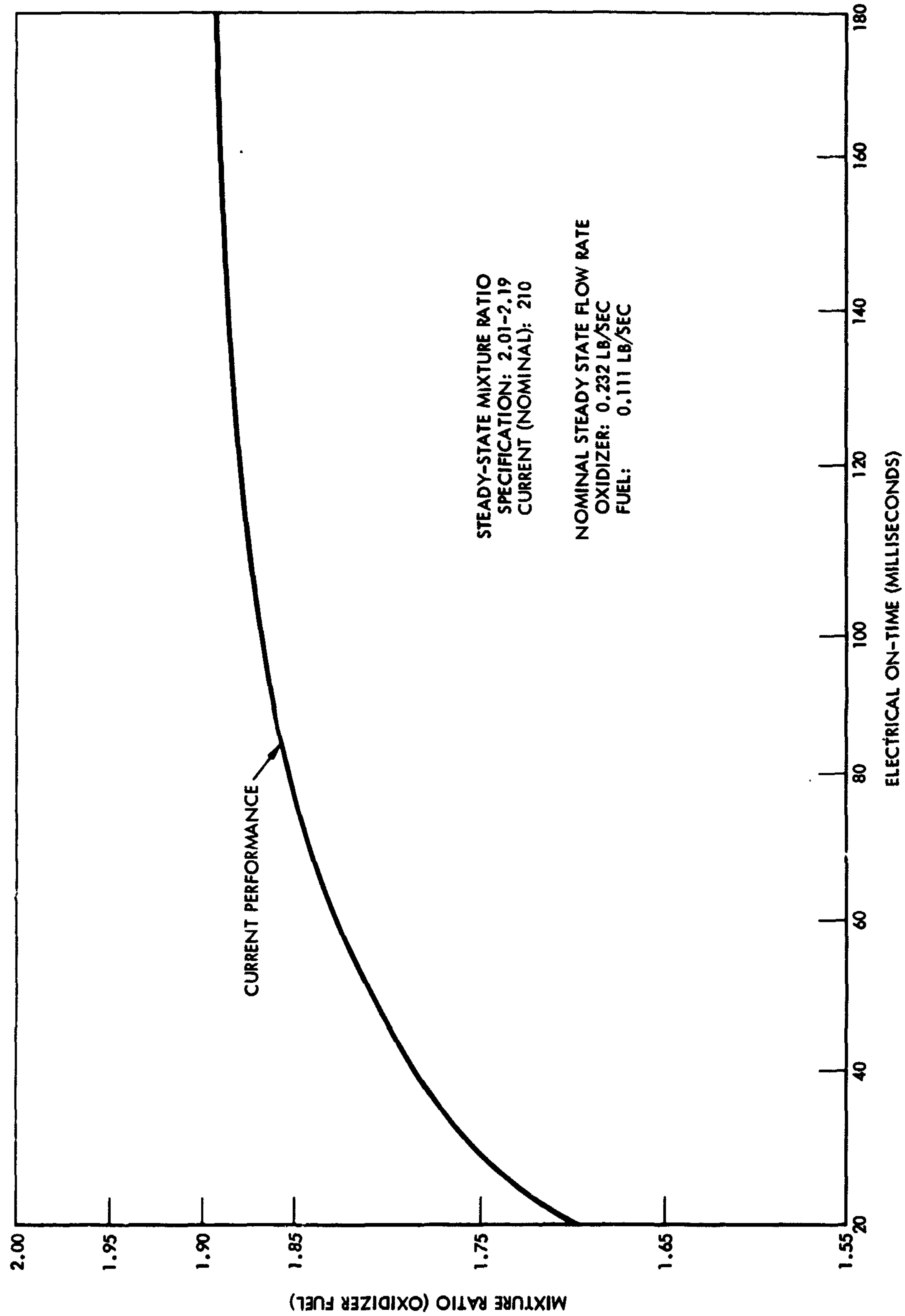


Figure 109. Mixture Ratio Versus Pulse Time for CM RCS Engine

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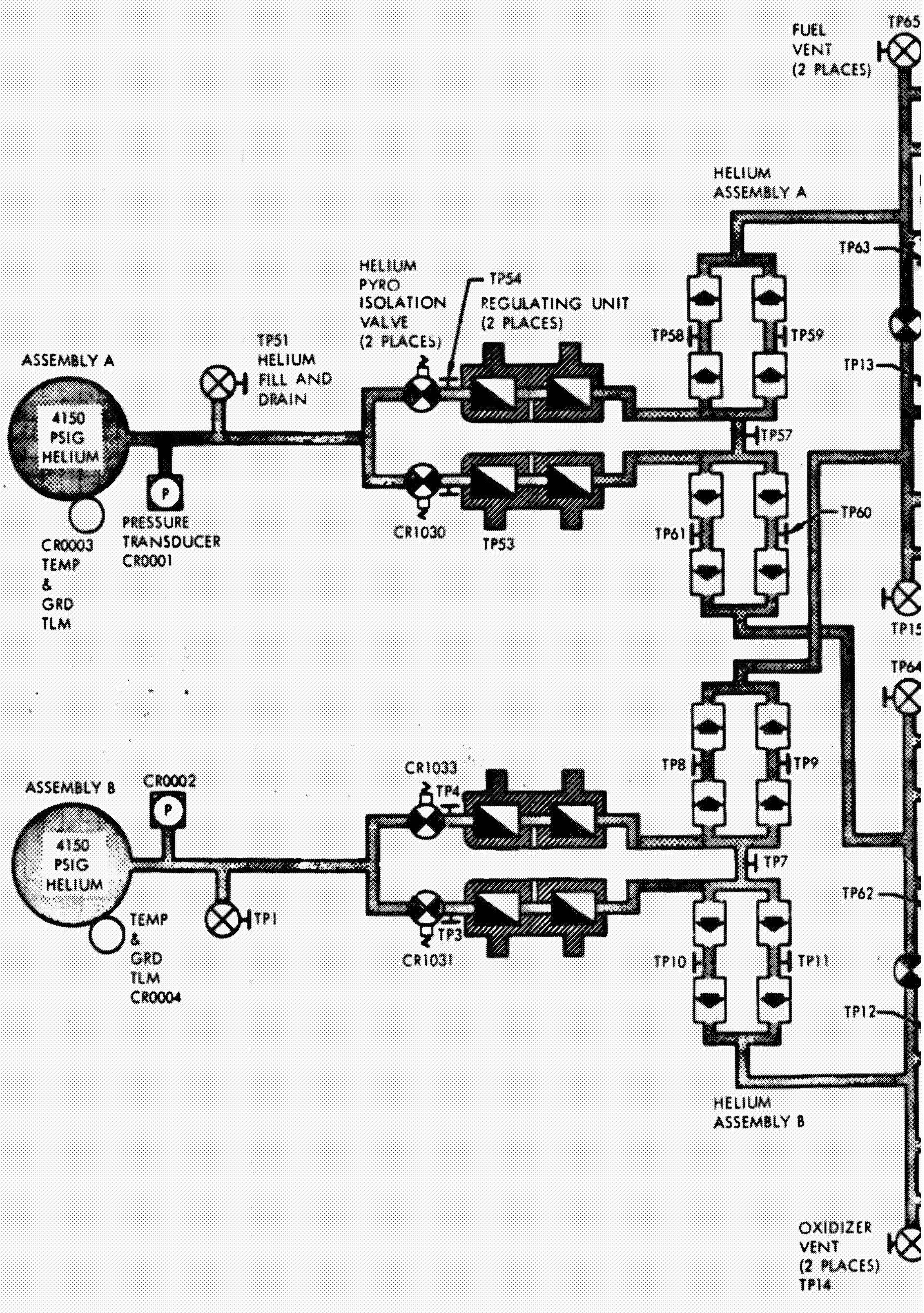

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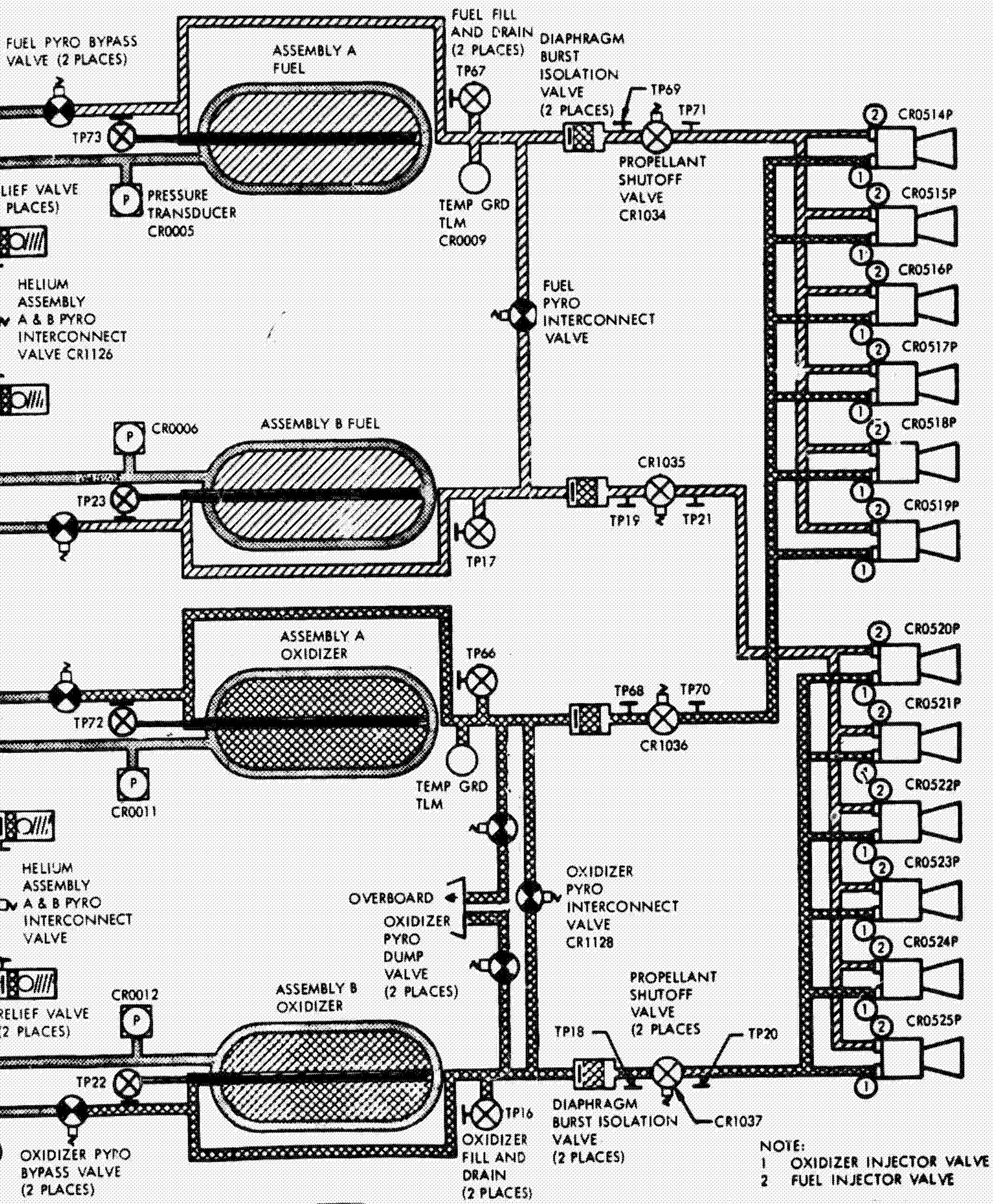

Figure 110. Schematic of Complete Command Module Reaction Control System

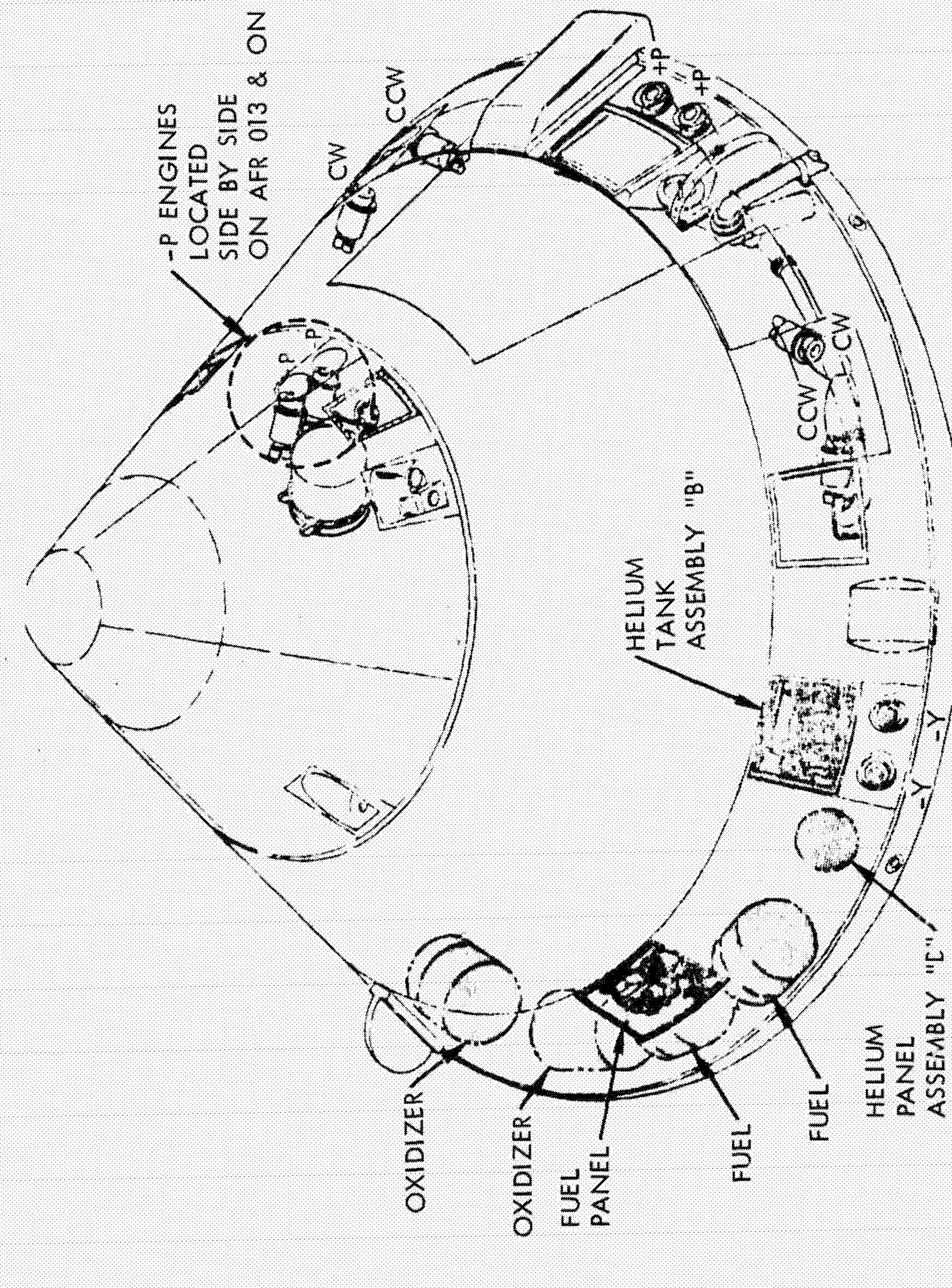

Figure 111. Perspective of Typical CM RCS Block II System

Table 78. Components of Apollo Command Module RCS

Component	Function	Type
Fuel tank	MMH storage	Positive expulsion with teflon bladder
Oxidizer tank	N_2O_4 storage	Positive expulsion
Propellant isolation valve	Isolate propellant until each entry, burst at 241 ± 14 PSID	Burst diaphragm
Flexible metal hose	Engine connect	Flex with dynatube fittings
Helium isolation valve	Retain He in pressure vessel	Explosive actuated
Propellant interconnect and dump valve	Isolate system A from B and dump	Explosive actuated
All other components are common to CM and SM systems and are covered in Table .		

Manufacturers and Availability Status. The Model 8250 system was manufactured by Bell Aerosystems Company for Lockheed Missiles and Space Division. The system is now out of production and its availability is unknown.

Space Programs. The Model 8250 system provided secondary propulsion to the Agena-Gemini Docking Target Vehicle.

Description of Functional Highlights. The Model 8250 system provided a vernier thrust capability in the forward axial direction so that the Agena GTV could assist in the rendezvous and docking maneuver with the manned Gemini capsule.

Functional and Physical Characteristics. Each Agena RTV was provided with two identical and independent Model 8250 systems. Each system included (see Figure 111) one 200-lb thrust chamber assembly, one 16-lb thrust chamber assembly, one bellows-type fuel tank, one bellows-type oxidizer tank, one helium tank, and associated pressurization components and valving. The 16-lb thrust chamber is described in Table 79, and the propellent tankage in Table 80. The system is shown schematically in Figure 112.

Integration Constraints. Unknown.

Application to RCM Laboratory. The Model 8250 system is designed to control one 200-lb and one 16-lb thrust assembly. In the RCM laboratory it would be necessary to control instead 12 or 16 16-lb thrust assemblies. The pair of propellant tanks in the Model 8250 system have about 92 percent the capacity of a single RCS quad, so provision of one to four pairs of tanks would be necessary, depending on the mission. The bellows-type tanks weigh more than comparable Apollo bladder-type tanks, but they should have the capability of withstanding longer mission durations. The other components of the Model 8250 system have not been studied, so their capabilities compared to comparable Apollo components are not known.

Table 79. Bell Model 8101/8250 Engine

Thrust	16 ± 1.6 lb at vacuum
Propellants	Mixed oxides of nitrogen (MIL-C-27408)/ unsymmetrical dimethyl hydrazine (MIL-P-25604)
O/F Weight Ratio	1.10 ± 3%
Rated Duration	2650 seconds design life
Engine Life	2 years maximum storage
Propellant Feed	Positive expulsion tanks
Thrust Vector Control	None
Planned Use	8101 - Agena Secondary Propulsion System, 8250 - Secondary Propulsion System, Agena-Gemini Target Vehicle
Status	8101 was qualified and flight tested on 4 successful flights. The 8250 has PFRT completed. Development: 8101 had preliminary design completed early 1961, PFRT July 1961 through September 1962; 8250 system had preliminary design completed January 1963, PFRT February 1964 through October 1964
Sponsoring Agency	Air Force/Lockheed for 8101 and 8250, 35 units of Model 8101 have been delivered. Production Contract for 8250 is AF 04(695)-545, for 18 production systems plus 2 spare thrust chamber units

Table 79. Bell Model 8101/8250 Engine (Cont)

Performance		Thrust Chamber*
Thrust, lb		16.0
Thrust Coefficient		1.746
Specific Impulse, lbf-sec/lbm		252.2
Characteristic Exhaust Velocity, ft/sec		4647
Chamber Pressure, psia		79
Flow Rates, lb/sec	Oxidizer	0.0332
	Fuel	0.0302

*At vacuum conditions

Thrust Chamber

Nozzle: Throat, D_t = 0.377 in., Exit, D_e = 2.832 in.,

A_t = 0.1118 sq. in. A_e = 6.28 sq. in.

Length, throat to exit = 3.715 in. Expansion area ratio, A_e/A_t = 55.6

Contraction 1/2 angle, = 40° Expansion 1/2 angle, β = 29°

Type: 80% Bell

Combustion Chamber:

Characteristic length, L* = 32 in. Chamber, D_c = 1.5 in.,

Chamber Temperature = 2400 F maximum A_c = 1.765 sq. in.

Injector: 6 doublets, impinging streams, no barrier flow

Table 79. Bell Model 8101/8250 Engine (Cont)

Cooling Technique: Radiation cooled nozzle and chamber

Chamber Construction: Tantalum + 10% tungsten, welded

Pressurization System

Nitrogen gas is used with stainless steel bellows for the 8250 and with a collapsing teflon bladder for the 8101

Weights = Model 8101 (in pounds)	Dry
Chamber and injector	2.2

Weights - Model 8250 (in pounds)	Dry	Wet
Overall system	129.2	303.8
Chamber and injector assembly	6.92	
Fuel tank	31.3	
Oxidizer tank	24.9	
Pressurizing systems	13.1	

Materials of Construction - 8101 and 8250

Propellant Tanks: Aluminum/teflon for 8101, Stainless steel, 347 bellows, 247 and A286 Shell for 8250

Valves: Aluminum/stainless steel

Injector: Pure tantalum

Chamber: Ta - 10 percent W with pyrochrome and aluminide coatings

Nozzle Extension: Pure tantalum

Table 79. Bell Model 8101/8250 Engine (Cont)

Operational Features	
Starting:	Hypergolic ignition of propellants
Restart Capability:	90 ~ after acceptance testing
Burning Time Limitations:	150 seconds maximum burn time per cycle
Power Supply:	28 watts at 28 volts dc
Reliability	
Failure rate per cycle:	1×10^{-6} (predicted)
References	
Data sheets supplied by Bell Aerosystems Company, September 1965	

Table 80. Principal Characteristics of Propellant Tankage Used in Model 8250 System

Item	Characteristic
Tanks	
Diameter	10 in. inside diameter
Length	
Oxidizer	33 in.
Fuel	42 in.
Material	A-286 steel
Thickness	0.018 in. wall
Working pressure	210 psi
Bellows	
Oxidizer	10 in. diameter x 93 convolutions
Fuel	10 in. diameter x 139 convolutions
Material	300 series stainless steel
Thickness	0.006 in.
Cycle life	200 expulsions
Operating temperature	0 to 100 F
Expulsion efficiency	98 percent

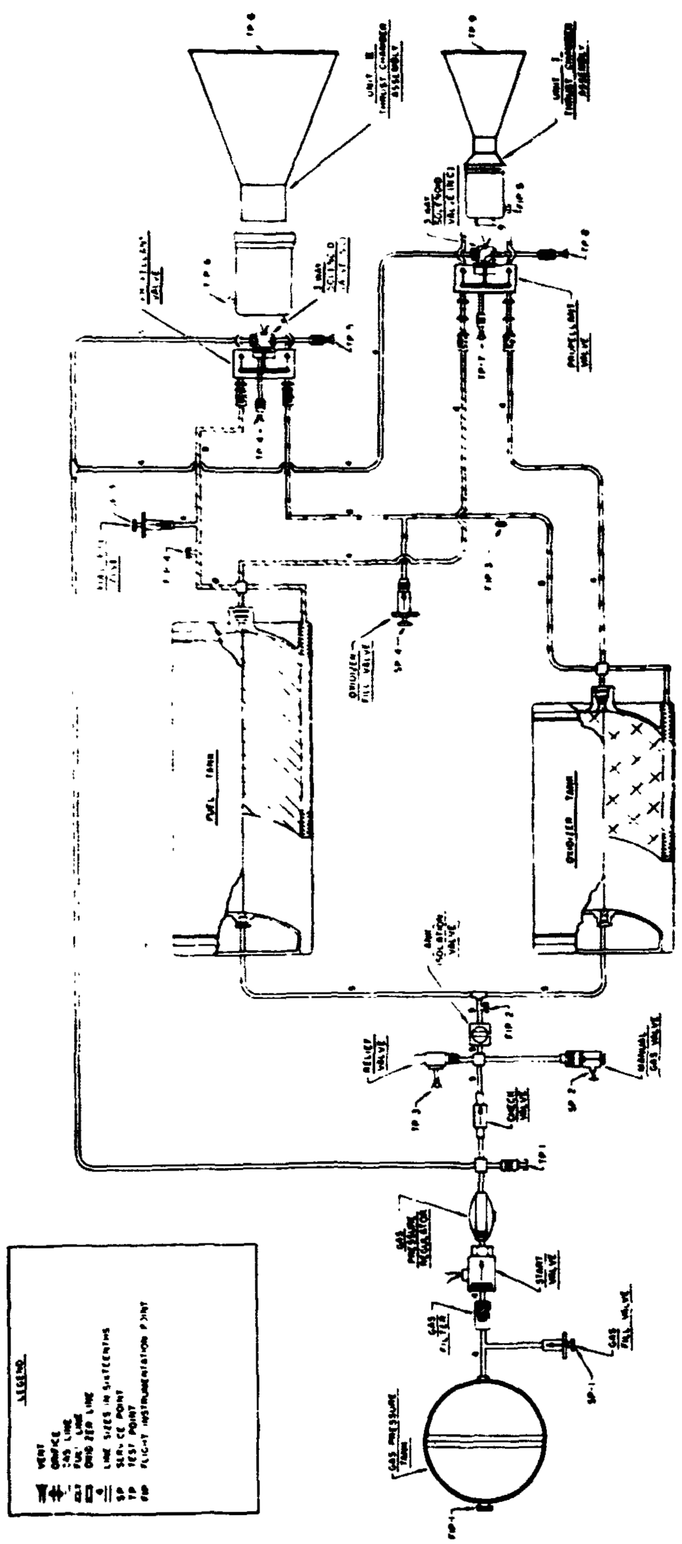

Figure 112. Secondary Propulsion System Model 8250 Schematic

29. Tank, O_2, Cryogenic Storage Subsystem. NAA S&ID Specification Control Drawing ME282-0046

30. Inverter, Power Static, 115/200 volt, 3-Phase, 400 Cycles per Second. NAA S&ID Procurement Specification MC495-0001

31. Zero-Gravity Positive Expulsion. Bell Aerosystems Brochure

APPENDIX: AVCO REPORT ON FEASIBILITY OF HEAT SHIELD REUSE

In support of this study, Avco Corporation performed a preliminary study to assess the feasibility of refurbishment of the Apollo command module heat shield.

This appendix presents the Avco report containing the results of their study. In addition to technical feasibility, production and cost aspects of refurbishment were covered.

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A PRELIMINARY ASSESSMENT OF THE REUSE AND REFURBISHMENT OF THE APOLLO COMMAND MODULE HEAT SHIELD

Prepared for

SPACE AND INFORMATION SYSTEMS DIVISION
NORTH AMERICAN AVIATION, INCORPORATED
Downey, California

AVSSD-0257-66-CR

by

O. K. Salmassy

14 October 1966

AVCO CORPORATION
AVCO SPACE SYSTEMS DIVISION
Lowell, Massachusetts

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ILLUSTRATIONS

ILLUSTRATIONS (Concl'd)

TABLES

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1.0 INTRODUCTION

This report is written in response to a request from the Advanced Programs Office of NAA/S&ID for an assessment of the refurbishment and reuse of the Apollo command module heat shield for low earth orbit Apollo Applications missions. Specific attention is given to defining major problems associated with demonstrating the feasibility of heat shield reuse and refurbishment. A conceptual development plan for refurbishment and associated costs are also presented.

This study is based upon the following guidelines and constraints:

1) Reuse will be limited to low earth orbir missions of 14-days duration.

2) Reliability requirements will remain unchanged from Apollo requirements.

3) Design performance criteria for refurbished vehicles will be identical with Apollo criteria.

4) Maximum utilization will be made of Apollo materials, processes and specifications.

5) NAA/S&ID will perform major vehicle disassembly and structural cleaning and repair prior to shipment to Avco/SSD.

6) Avco/SSD will perform refurbishment of all heat shield panels and components for which it has Block II fabrication responsibility.

7) Vehicles recovered from earth orbital as well as lunar missions will be refurbished.

2.0 FACTORS AFFECTING FEASIBILITY OF REFURBISHMENT OF APOLLO C/M HEAT SHIELD

Post-recovery examination of Apollo C/M 009 and C/M 011 following entry from low earth orbit has indicated that the heat shield ablator survived entry quite well with apparent insignificant degradation in many areas. In other areas the depth of degradation or char, appears relatively shallow. In still others the local degradation is significant. Since the Apollo heat shield is designed for entry from lunar missions, this result was expected. The degree of degradation of the ablator from entry at lunar velocity, however, is predicted to be significantly greater. The range of true depths will not be known, however, until the recovery of C/M 017 and C/M 020.

In view of the wide range of local degradation anticipated, three conceptual approaches may be feasible for local refurbishment:

a) complete reuse of recovered ablator,

b) removal of char and reuse of uncharred ablator,

c) complete removal of original ablator, charred and uncharred, and application of new ablator.

(The reuse of heavily charred ablator material is not considered feasible due to its poor mechanical integrity).

It is conceivable that all three or combinations of these approaches could be employed on a single vehicle. Table 2-1 indicates the conceptual utilization of these refurbishment approaches. The feasibility of the first two of these refurbishment approaches has not been demonstrated. Approach C, complete replacement, is essentially equivalent to a standard Class 3 repair technique utilized currently on flight hardware. Its feasibility has been demonstrated on C/M 009 and C/M 011. As a result, primary attention has been given here to the problems associated with demonstrating the feasibility of Approaches A and B. The feasibility of these two approaches will depend heavily on the C/M's initial mission - being much more probable for initial earth orbital missions than for initial lunar missions because of the latter's greater char depth. The initial entry trajectory will also influence char depth and thereby the local refurbishment approach. Such considerations are indicative of a major problem in the selection of the refurbishment approach, i.e., the predetermination of char depth. The nature and influence of this problem and other major factors affecting refurbishment are discussed in the following sections:

TABLE 2-1

CONCEPTUAL UTILIZATION OF REFURBISHMENT APPROACHES

Approach	Uncharred Areas	Charred Areas	
		uncharred greater than required	uncharred less than required
Class A: complete reuse	X		
Class B: reuse uncharred		X	
Class C: complete replacement	X	X	X

2.1 CHAR DEPTH

The predetermination of local char depth, or the depth in the ablator to which the char has penetrated, plays a major role in the selection of refurbishment approach, i.e., unless there has been a decision to completely replace old ablator with new regardless of char depth. The depth of char will be a strong function of the C/M's initial mission and entry trajectory. For example, for the aft compartment char depth for earth orbit entry may average on the order of 25% of total thickness, whereas, for lunar entry the char depth may average on the order of 50%. On the basis of the meager post-flight data available to Avco/SSD on C/M 011, it would appear that these char depth estimates may be conservative, i.e., the actual post-flight uncharred material may be larger than predicted.

A more explicit characterization of earth orbital char depth and its distribution cannot be obtained until such time as appropriate post-flight test data on C/M's 009 and 011 become available. In order to determine the feasibility of Class A and B refurbishments of heat shields which have undergone prior earth entry, analyses of post-flight data on C/M's 009 and 011 should be conducted. This is required to assess the adequacy of analytic methods for predicting char depths and to obtain a realistic appraisal of char depth distribution and of local anomalies. Determination of the feasibility of refurbishment of lunar heat shields will require similar study of AFMS 017 and 020.

An important factor which enters into the overall problem of feasibility is the definition of char depth, i.e. the criteria used to determine the depth of degraded and nonreusable ablator. As illustrated in Figure 2-1, a definition of "char depth" corresponding to minimum density instead of undegraded density might involve a difference of as much as 0.2 inches in material available for reuse. It is conceivable that judicious selections of this definition might permit a Class B refurbishment where a Class C refurbishment would otherwise be required. It is clear, however, that knowledge of the ability to reuse partially degraded ablator is required to demonstrate feasibility of Class A and B refurbishments which reuse partially degraded ablator.

As shown in Figure 2-1, char depth may be defined in terms of ablator density. It also appears possible to define char depth in terms of the maximum temperature of in-depth exposure during entry. Neither of these techniques is really practical for recovered flight vehicles, however, because of the complexity and cost of obtaining density and temperature data. Two other techniques that have been used are color and electrical resistivity; however, too little experimental data are available to determine if these techniques are sufficiently consistent to yield reliable data for machining. Color recognition and definition are highly subjective and hence, subject to large observer error, particularly in deep chars generated over long heating periods. Electrical techniques have been used successfully in isolated analyses of test chars, but are unproven for production applications. The net result is that no technique can be shown at this time to be both feasible and practical.

On the basis of the information available at this time, an NDT technique based on the measurement of the dielectric properties appears most feasible. In addition, because of extensive experience on Apollo in the use of this technique for determining ablator moisture content (Figure 2-2), dielectric probes, necessary instrumentation, and basic technology appear to be available. The dielectric approach, using these previously developed coplanar probes, should permit identification of surface char and char-virgin transition. The transition zone, being a region where density and resistivity change markedly, should provide a wide range of dielectric properties, and hence, an excellent thickness sensitivity. The major task remaining appears to be demonstration of the feasibility of reduction of this technique to production use.

In summary, assessment of the following factors is necessary to demonstrate the feasibility of Class A and Class B refurbishment:

1) Acceptable degree of degradation for reuse

2) Char depth measurement techniques for production use.

Char depth is not a problem in Class C refurbishment.

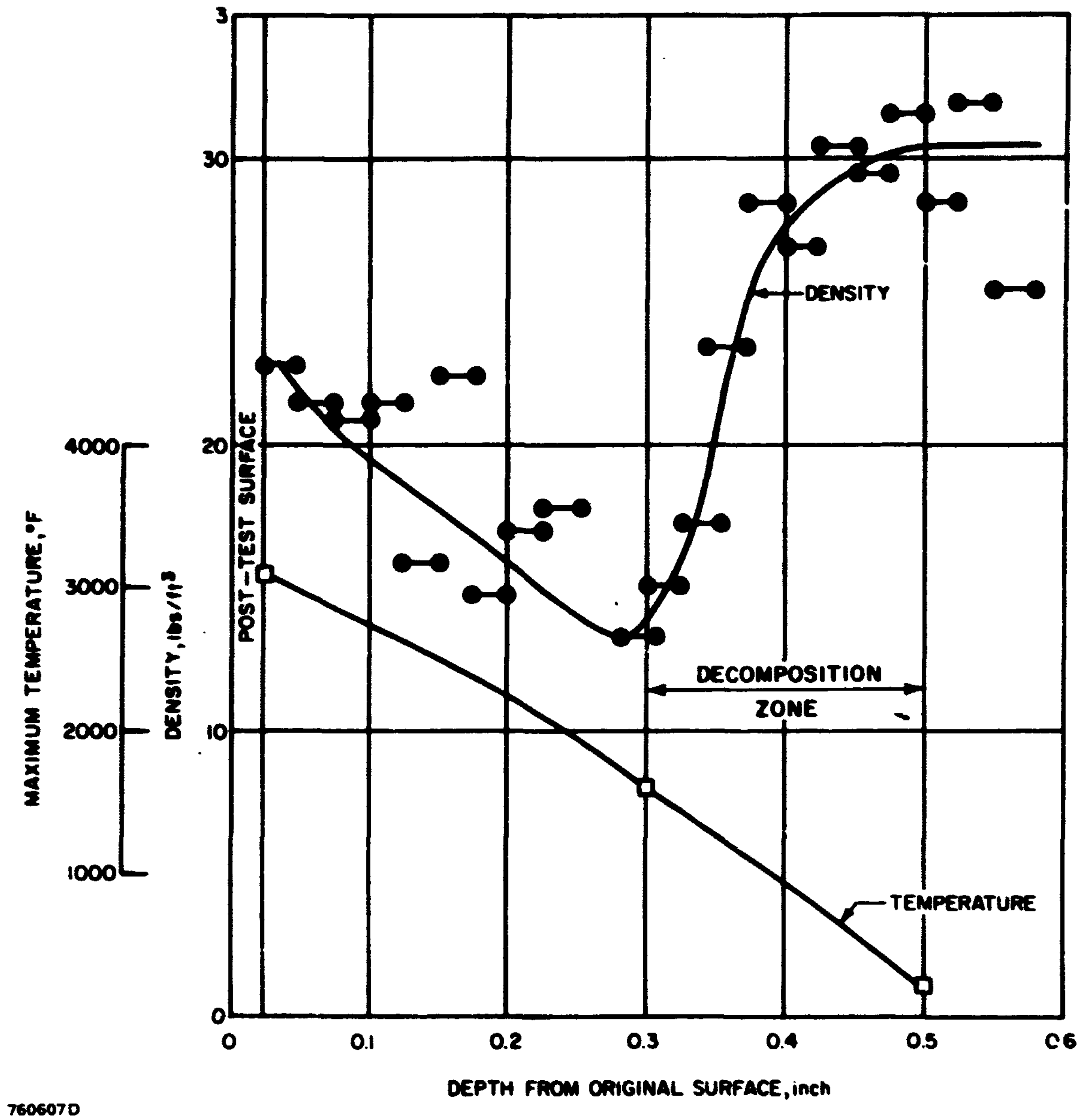


Figure 2-1 DENSITY PROFILE IN CHARRED APOLLO ABLATOR SAMPLE

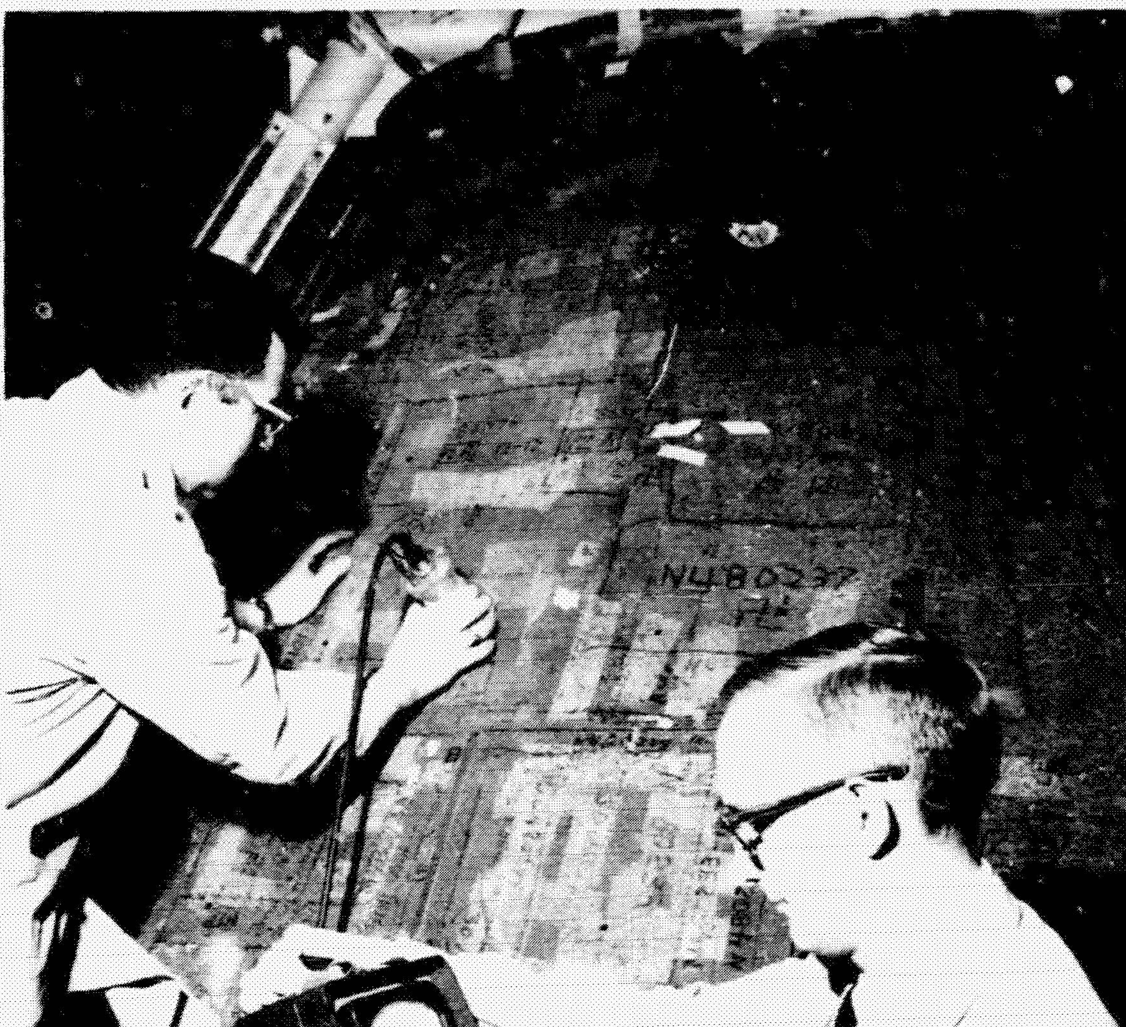


FIGURE 2-2. APPLICATION OF DIELECTRIC NONDESTRUCTIVE TECHNIQUES TO MEASUREMENT OF APOLLO ABLATOR MOISTURE CONTENT

2.2 PERFORMANCE OF USED ABLATOR

Class A and Class B refurbishment require the reuse of ablator materials which have been subjected to prior entry thermal and structural loads. Evaluation of the adequacy of performance of reused materials is necessary, therefore, to the demonstration of feasibility of Class A and B refurbishments. The feasibility of Class C refurbishment is not affected by such considerations.

The initial Block II entry will induce thermal gradients through the uncharred ablator material. If this uncharred material is to be reused for a second entry, the effect of this previous temperature history on subsequent thermal and structural margins of safety must be understood. For Class A and Class B refurbishment, depending upon the definition of char depth or, conversely, the definition of degradation acceptable for reuse, the maximum exposure temperature could conceivably range from 250°F to 1000°F. The upper temperatures will be achieved (for short times) on ablator materials near the char-virgin interface. The lower temperature limit would correspond to material near the bondline which did not achieve an entry temperature in excess of the initial 250°F post-cure temperature. Because of the potential variations of initial Block II entry trajectories as well as potential variations of earth orbit trajectories after refurbishment, it is not possible at this time to define the spectrum of material histories to be encountered by "reusable" material. The definition of this spectrum can be acquired only after the variations in entry missions have been explicity defined along with the definition of acceptable char depth. It does appear, however, that it should be possible to bound the thermal performance problem by evaluating residual, uncharred materials exposed to the upper temperature bounds. Currently, no data are available on the effect of transient, elevated (above cure temperature) thermal exposures approaching 1000°F. Data are available, however, for exposures to 350°F. For exposures as long as 100 hours at 350°F, the ablation performance of the Apollo ablator appears unaltered as shown in Table 2-2. No alteration of char depth was noted.

Several factors that will affect the structural performance of used ablator must also be investigated before the feasibility of reuse can be determined. The structural performance of the ablator on Apollo is measured in terms of margins of safety for cracking and for delamination from the substructure. The former, most critical when the ablator is cold, is primarily a function of the allowable mechanical strain of the ablator, the difference between the free thermal contraction plus additional shrinkage of the ablator, and the thermal contraction of the substructure referenced to the zero stress temperature. This margin of safety can readily be calculated once the ablator properties - modulus, ultimate strain allowable, coefficient of thermal expansion, and additional shrinkage, are known. The margin of safety for delamination is a function of the local curvature of the substructure at a crack: It can be computed once the ablator properties and the allowable curvature are known.

TABLE 2-2

ABLATION PERFORMANCE DATA ON APOLLO ABLATOR SUBJECTED TO ELEVATED THERMAL EXPOSURE (350°F FOR 100 HOURS)

Sample History	H_s (Btu/lb)	$\dot{q}_c$ (Btu/ft^2-sec)*	Time (sec)	Specific Gravity	Length Ablated	Surface Temp (°K)
Elevated Exposure after Cure	10,600	108	240	0.466	0.457	1975
	10,000	116		0.466	0.513	2015
	10,200	114		0.462	0.478	2005
Control: Std. Apollo	10,400	109	240	0.462	0.443	2008
	10,500	110		0.455	0.4[illegible]	2025
	10,200	112		0.462	0.479	2005

*Impact pressure, 7.0 mm Hg

Then, to calculate structural margins of safety requires knowledge of "reusable" ablator properties. In a similar sense, the residual margin of safety associated with prior repairs made to the original Block II heat shield would be required.

When an ablator is subjected to reentry heating, every portion of it will experience some time-temperature heating history that will affect its subsequent mechanical properties. For instance, it is known that the unrestrained ablator will shrink when exposed to elevated temperature and that this phenomenon is time-temperature dependent. Figure 2-3 depicts unrestrained behavior as a function of temperature that has been increased and decreased at a uniform rate of 3°F per minute. The nearly linear behavior up to 350°F is reversible, but the large shrinkage that occurs at higher temperatures is irreversible. Similar, more pronounced behavior has been noticed when the temperatures have reached 800°F. The ablator on the vehicle is restrained, however, and some test results have indicated that restrained ablator will creep and partially relieve the induced strains. Furthermore, as is shown in Figure 2-4, shrinkage at a given temperature is time dependent. Thus, had the heating rates in Figure 2-3 been increased to those of an actual reentry, the irreversible shrinkage would have been far less. This is further borne out by the appearance of restrained ablator samples that have been subjected to simulated reentry radiant lamp testing wherein the char layer is extensively cracked but none of the cracks appear to extend into the virgin material.

In addition to affecting ablator shrinkage, prior exposure to elevated temperatures will probably have an effect on ultimate allowable strains and moduli of elasticity. All these time-temperature effects on ablator properties and shrinkage must be investigated before the feasibility of reusing ablator can be determined.

It is fairly evident at this time that a used heat shield would not be able to survive as cold a temperature as before entry without cracking because of the additional shrinkage strains; but just what minimum temperature it could survive is unknown.

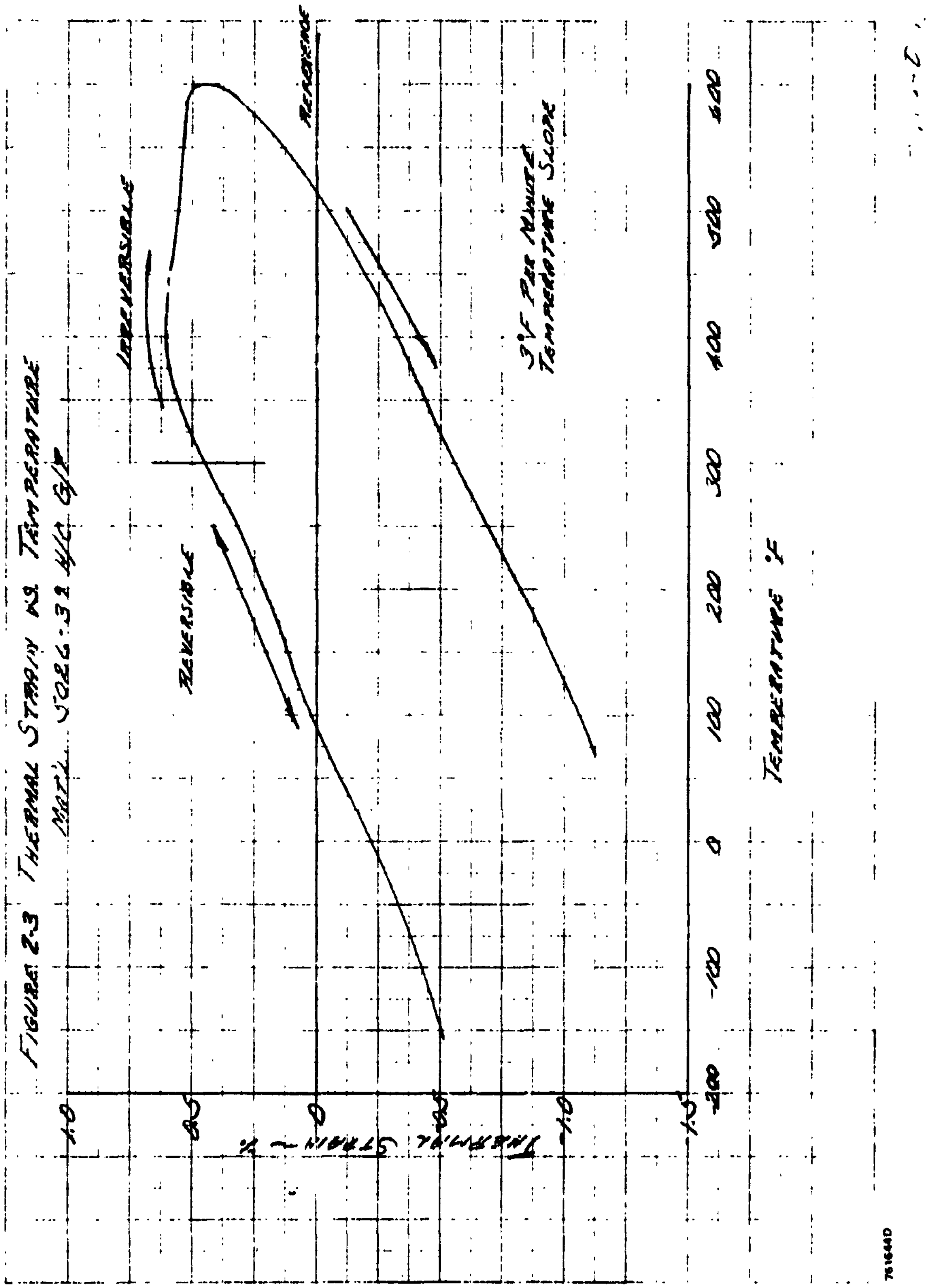


Figure 2-3 THERMAL STRAIN VERSUS TEMPERATURE (MATERIAL 5026-39 H/C G/P)

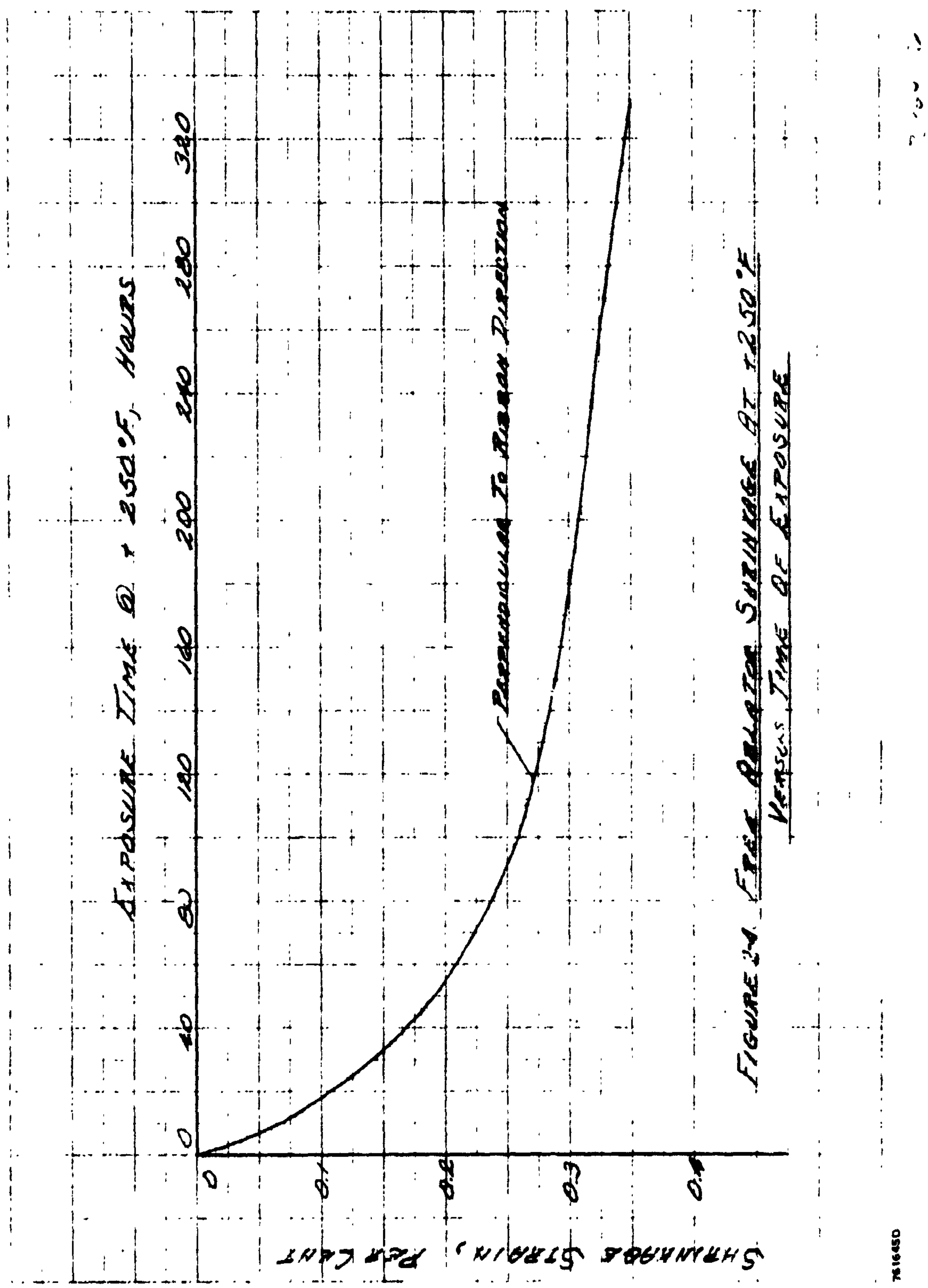


Figure 2-4 FREE ABLATOR SHRINKAGE AT +250°F VERSUS TIME OF EXPOSURE

There is some reason to believe, however, that prior thermal exposure may not be as detrimental to structural performance as might at first be supposed. In the early phases of Apollo C/M development, when there was no boost cover, the ablator material was required to undergo significant ascent heating. This heating resulted in charring and degradation of the outermost 0.1 to 0.2 inch of ablator. The degraded heat shield was then required to be capable of sustaining -260°F cold soak, 250°F hot soak, and reentry.

Heat shield panels were subjected to ground tests which sequentially exposed the ablator to simulations of these environments. For example, the test panel shown in Figure 2-4A was subjected to simulated ascent heating under radiant lamps. (The deep black color of the surface is due partly to the application of carbon black to improve absorptivity under the lamps.) A light surface char and subsurface degradation were created by this exposure. The panel was then subjected to a cooling rate simulating spaceflight temperature predictions and after 54 hours reached -260°F ± 2°. It was soaked at this temperature for two hours and then raised to room temperature. Figure 2-4B shows the panel after cold soak. The panel was then hot soaked by raising the surface temperature to between 400 and 500°F until the bond line reached 250° to 260F. It was held in this condition for 45 minutes with a flow of CO_2 across the surface. The maximum surface temperature reached during this period was 470°F. The panel was then subjected to simulated reentry heating in the radiant lamp facility. The test panel after heating is shown in Figure 2-4C. No cracks were found after cold soak or reentry heating.

In these tests, the ascent heating and consequent surface charring and degradation were found to have no apparent detrimental effect on subsequent cold soak and reentry performance. On the basis of these tests, it would appear that there may be reason to believe that reuse of degraded ablator in Class A and B refurbishments may prove structurally feasible.

An important guideline that directly affects the local feasibility of Class B refurbishment, and thereby total vehicle refurbishment cost, is the current requirement to use Apollo design criteria on allowable backface temperature for the reuse Earth orbit mission. Since total ablator thickness for the reuse mission is a strong function of allowable backface or bondline temperature, any relaxation of this criterion would significantly reduce the local, uncharred ablator required for Class B refurbishment. Relaxation of this criterion does not appear to affect anticipated thermostructural performance. In fact, the resultant reeuced ablator thicknesses would be expected to reduce any deleterious effects of cracking by reducing the degree of crack opening and by reducing delaminating stresses.

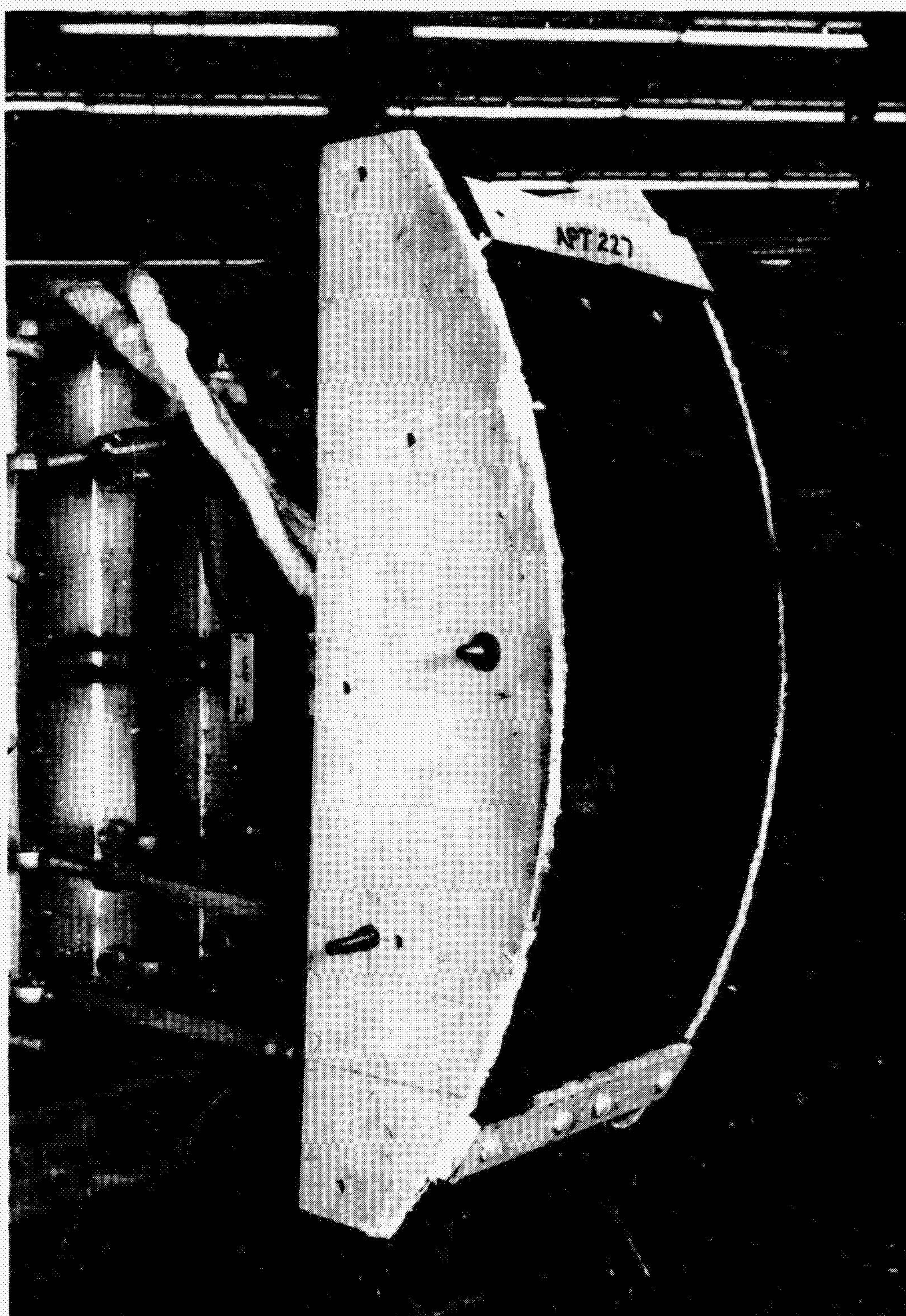


FIGURE 2-4A. TEST PANEL AFTER SIMULATED ASCENT HEATING

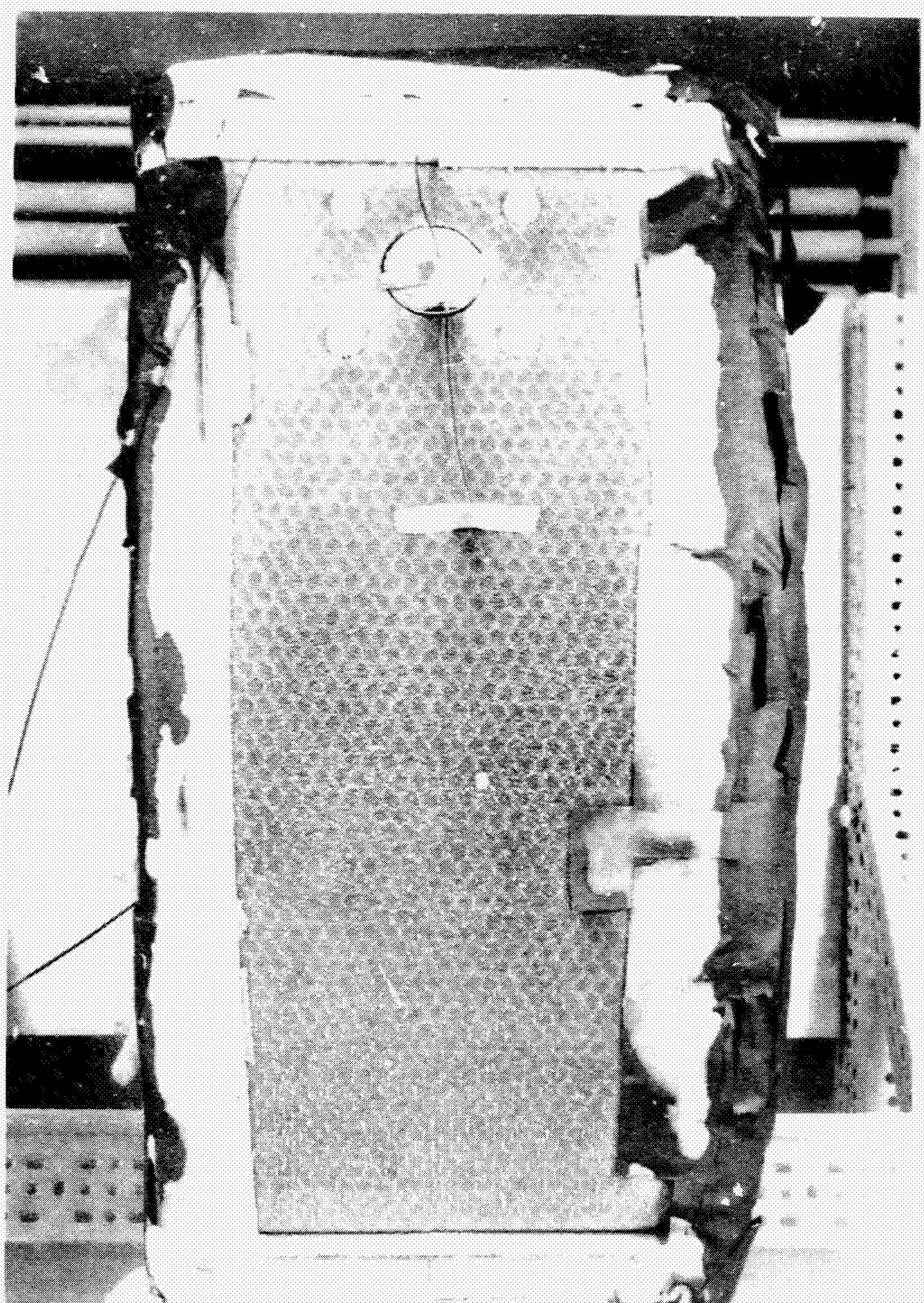


FIGURE 2-4B. TEST PANEL AFTER SIMULATED COLD SOAK

FIGURE 2-4C. TEST PANEL AFTER SIMULATED REENTRY HEATING

In summary, assessment of the following factors is necessary before feasibility of reusing ablator can be determined:

1) The effect of prior exposure to appropriate time-temperature histories on the ablator mechanical properties of modulus, ultimate strain, coefficient of thermal expansion, and shrinkage, must be identified.

2) The effect of such exposure on the substructure bending curvatures that would cause ablator and repair delamination must be found.

3) The effect of prior exposure on the aerothermodynamic performance of used ablator.

2.3 SALT WATER IMMERSION

Salt water immersion of the ablator will occur following entry of the C/M from its initial Block II mission. Post-recovery examination of C/M's 009 and 011 has indicated that heavily charred areas such as those to be expected on the aft compartment and windward portions of the crew compartment will absorb salt water in significant quantities. (Data on actual weights absorbed are not available.) Cracks or separations which naturally occur along honeycomb-ablator interfaces in the char together with the high char porosity may permit penetration of salt water throughout the greater portion of the char. Whether or not salt water penetrates completely through the char or into the virgin ablator is unknown.

A non-destructive test technique will be required to determine the presence of absorbed water in uncharred ablator for Class A and Class B refurbishments. A demonstrated nondestructive test method, based on dielectric techniques, is available for measuring the presence of non-saline water; but no technique has been demonstrated for measuring absorbed salt water.

It appears, at this time, based on Apollo specifications, that moisture retained in uncharred ablator in excess of 3% by weight would require removal by drying. No data are available on the effect of known quantities of absorbed water on ablator properties. Data available, however, on the effects of exposure to 90% relative humidity for periods up to 30 days indicate no significant effect on ablator properties, as shown in Table 2-3.

If recovered, uncharred ablator is found to contain greater than 3% water by weight, and if Apollo specifications imposed for weight control are applied, then the recovered compartments would require drying to remove excess water. If salt water drying procedures must be developed, then a simultaneous determination of the effect of these processes and of retained crystalline salt on ablator thermal and mechanical performance would be required. Such effects are currently unknown. It is important to note that it should not be apriori assumed that absorbed salt water or retained salt are detrimental to ablator performance.

TABLE 2-3

EFFECT OF 90-PERCENT RELATIVE HUMIDITY EXPOSURE FOR 30 DAYS ON APOLLO ABLATOR PROPERTIES*

Property	Test Temperature (°F)	Property Value (Avg.)	
		Test Group (Exposed)	Control Group (Unexposed)
Modulus of elasticity, 10^6 psi	70	0.071	0.075
	-150	0.090	0.11
Ultimate tensile strength, psi	70	348	525
	-150	500	478
Ultimate tensile strain, percent	70	0.54	0.77
	-150	0.54	0.48
Thermal Strain	185 to 70	0.16	0.13
	70 to -150	0.42	0.40
	185 to -150	0.58	0.53

*Estimated 10-percent water content by weight.

Heat shield materials other than the main ablator material, e.g., edgemember, bolt plug sleeve, and gasket materials, which generally are impervious to salt water, are not expected to be affected by immersion.

The impact of NAA/S&ID cleaning and refurbishment brought on by immersion of substructure elements such as aft compartment shear pad Marinite is currently unknown. If such refurbishments can be carried out without disturbing ablator - or edgemember - substructure interfaces, then Class C refurbishment of these interfaces may be minimized.

Class C refurbishment appears feasible from the immersion standpoint. Unless salt water penetrates the HT-424 tape and cannot be neutralized or flushed and consequently endangers substructure face sheet integrity through corrosion or endanger bond integrity,Class C refurbishment based upon removal of all ablator to the HT-424 layer should be feasible. Generally, application of the HT-424 primer in combination with the flow of primary HT-424 adhesive would be expected to shield the steel face sheets from salt water.

In summary, assessment of the following factors is necessary to demonstrate the feasibility of Class A and Class B refurbishments:

1) Maximum allowable retained salt water or retained salt for acceptable thermal and structural performance.

2) Ablator drying processes for acceptable thermal and structural performance.

No immersion factors appear to impair the feasibility of Class C refurbishments.

2.4 DESIGN CONSIDERATIONS

Refurbishment of the Block II Apollo heat shield for reuse on earth orbit missions will require generation of new design thicknesses. Since design techniques will have been verified by flight tests of C/M's 017 and 020, feasibility of design for reuse will have been verified. Furthermore, with appropriate post-flight analyses, it should be possible to design for the reuse mission with known margins and possibly reduced thicknesses over those currently obtainable.

Since each reuse of a flight C/M will require fabrication of a new forward compartment, it is assumed that new forward compartment ablator thicknesses will be generated for the reuse earth orbit missions.

Refurbishment class selection will require consideration of X- and Z-axis c.g. effects.

In general, reduced ablator thicknesses and refurbishment requirements will involve all major Block II heat shield components and a large number of Avco/NAA design interfaces. Some of the major effects of refurbishment and redesign for reuse are described in the following sections.

In summary, no design factors have been identified which constrain the feasibility of Class A, B, or C refurbishments.

2.4.1 Aft Compartment: Shear and Compression Pads (Figure 2-5 and 2-6)

All aft compartment shear and compression pads will require replacement, or class C refurbishment, due to the severity of heating experienced, even from orbital entry. Load transmission requirements permit no degradation of mechanical properties from heating. Reduction in local main ablator thickness for the reuse mission will, in turn, cause the Block II pads to protrude if they are to accept the current S/M interfaces. Reduction of pad thicknesses will require re-verification of thermal and structural performance and redesign of the S/M pads. Maintenance of Block II design appears preferable and feasible if the resultant perturbed local heating is shown to be insignificant. Redesign of local gasketing does not appear to be required.

2.4.2 Aft Compartment: RCS Oxidizer Dump (Figure 2-5)

This part is supplied, installed and gasketed by NAA/S&ID. Avco/SSD prepares edgemember closeout.

Reduction of local main ablator thickness will require design change of dump ablator plugs to prevent perturbed heating. No local redesign of main ablator except thickness is anticipated for Class B or C refurbishment.

2.4.3 Aft Compartment: RCS Fuel Dump (Figure 2-7)

This part is supplied, installed, and gasketed by NAA/S&ID. Avco/SSD prepares edgemember closeout.

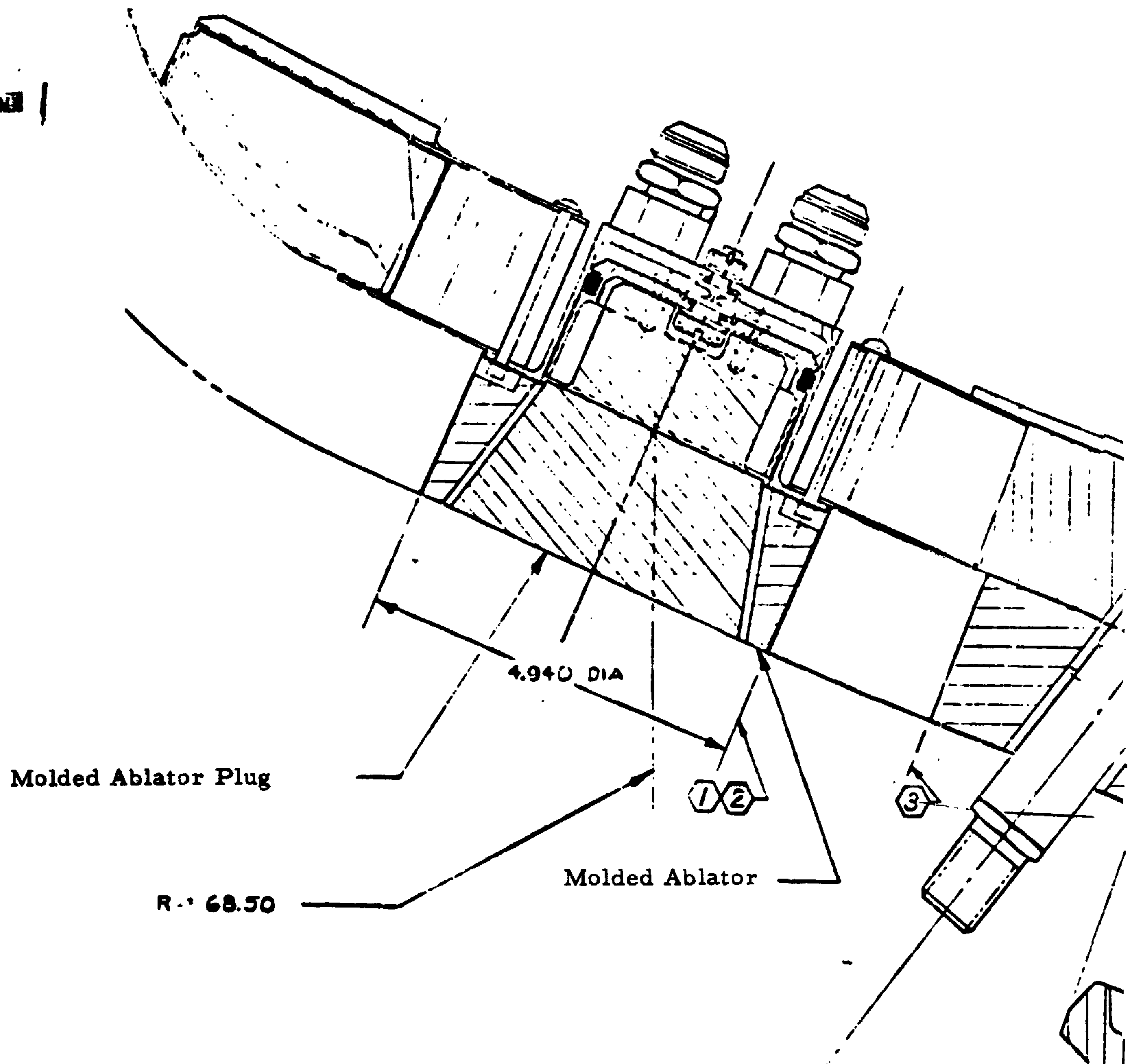
No redesign of local main ablator except thickness is anticipated for Class B or C refurbishment. Reduction of NAA plug thickness is anticipated to obviate perturbed heating effects.

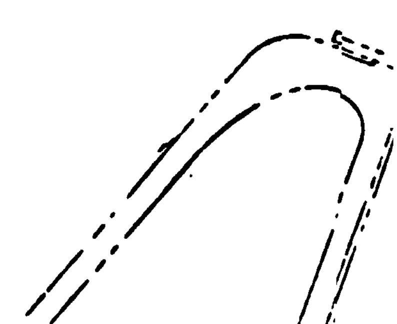
2.4.4 Aft Compartment: C-Band Antenna (Figure 2-8)

AFM 101 and 102 only. This part is furnished, installed, and gasketed by NAA/S&ID.

No redesign of local main ablator except thickness is anticipated for Class B or C refurbishment. NAA/S&ID removal of C-Band antennas is expected for reuse missions. Replacement with a plug in AFM 101 is anticipated.

FOLDOUT FRAME





1. NAA-AVCO Interface
2. Nonadherent Surface.
   Joint & Gasket design by AVCO.
   Fabricated by NAA.
3. Nonadherent Surface.
   Joint & Gasket Design by AVCO.
   Fabricated by AVCO.

AFT COMPARTMENT SHEAR PAD
and RCS OXIDIZER DUMP

FOLDOUT FRAME 2

FOLDOUT FRAME

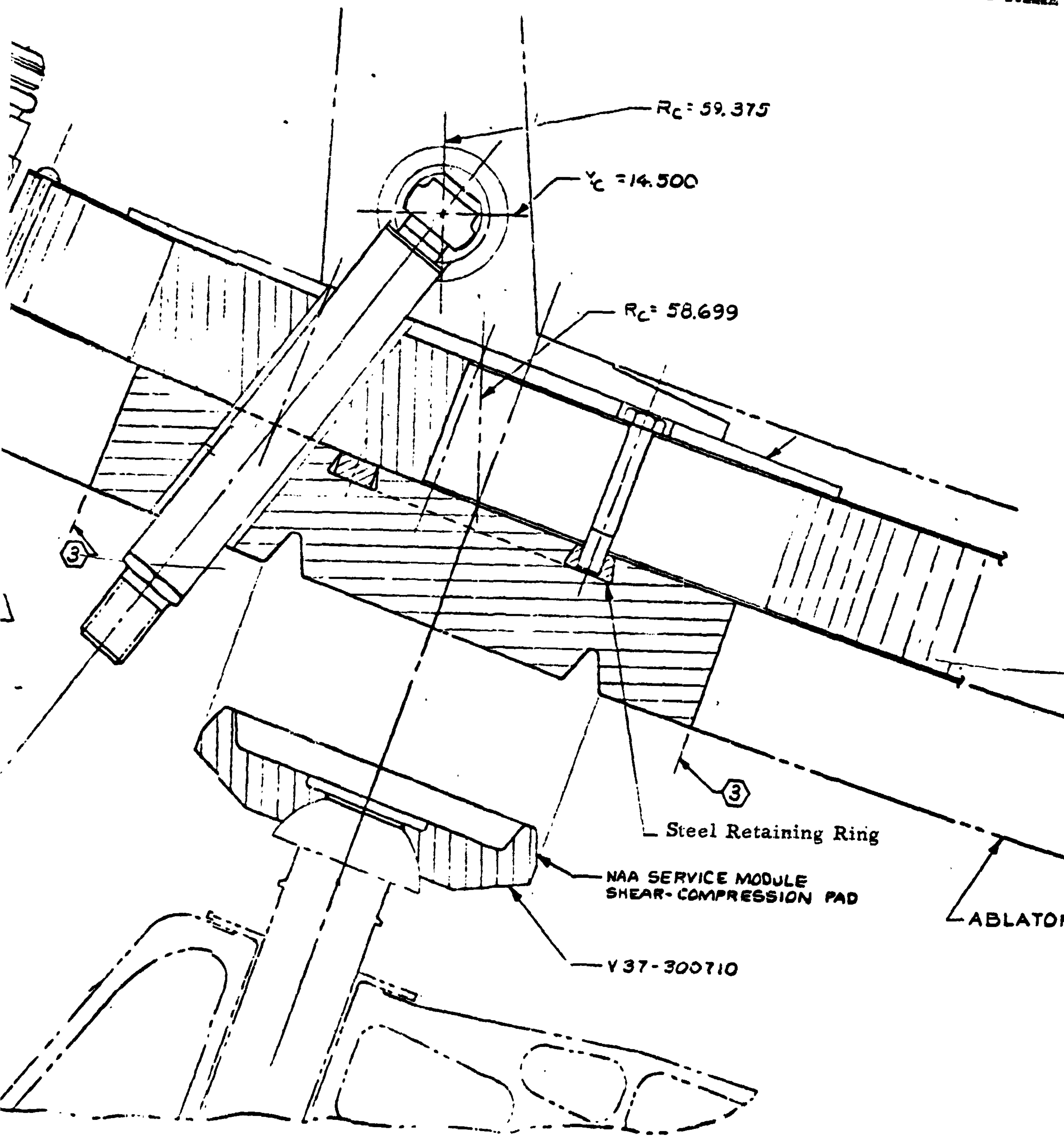
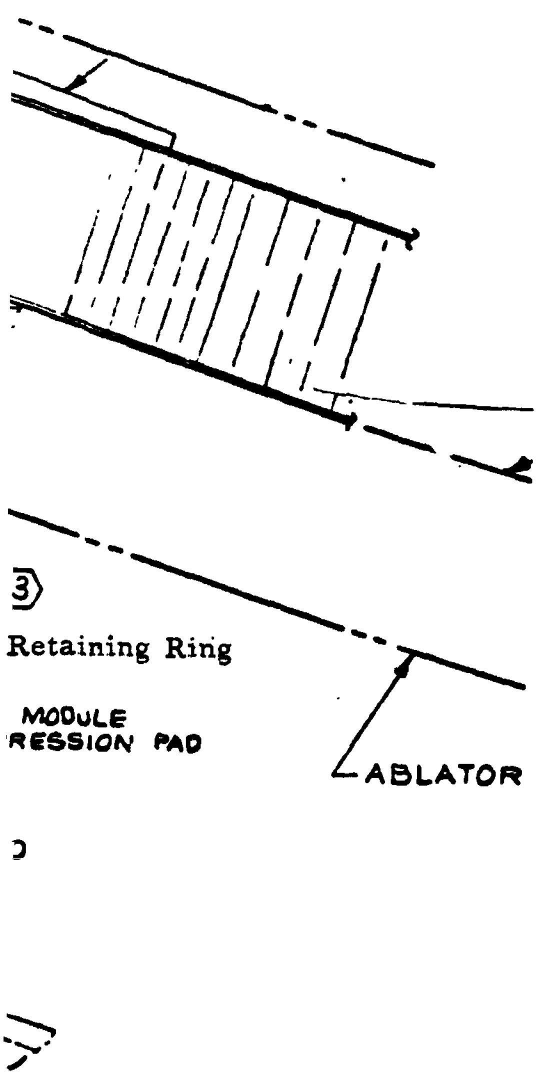


Figure 2-5 AFT COMPARTMENT SHEAR PAD AND RCS

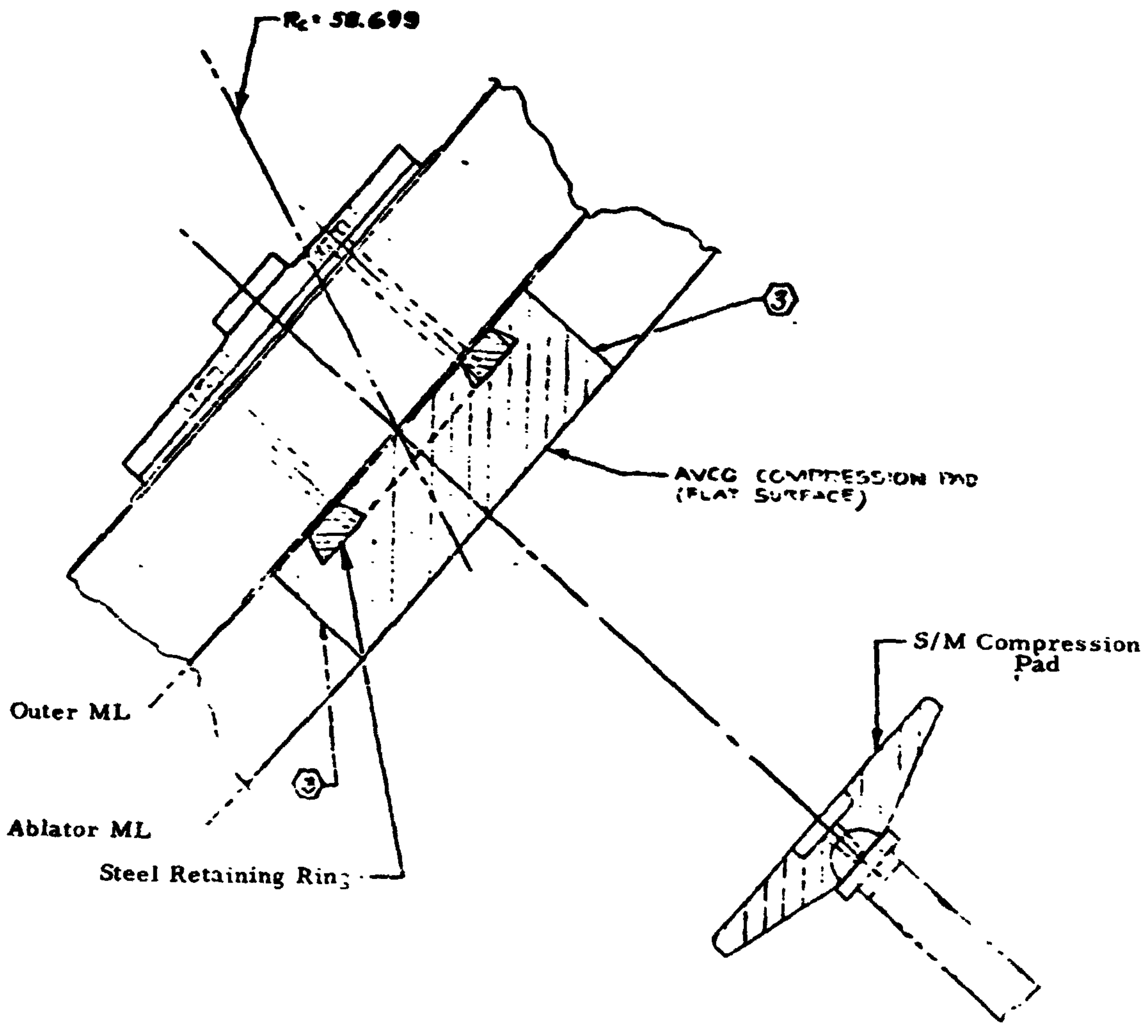
FOLDOUT FRAME 3

375



AFT COMPARTMENT SHEAR PAD AND RCS OXIDIZER DUMP

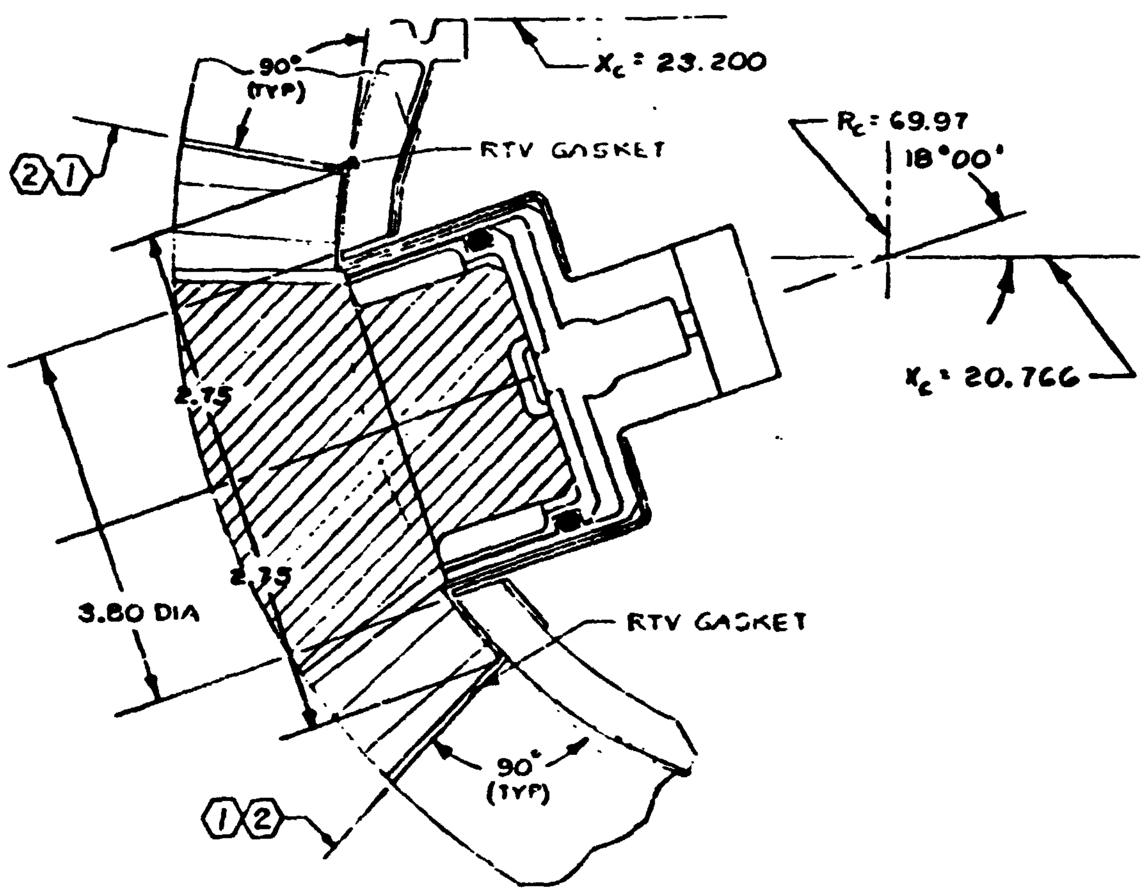
A-31, A-32



(3) Nonadherent surface.
Joint & Gasket design by Avco.
Fabricated by NAA.

7616470

Figure 2-6 AFT COMPARTMENT COMPRESSION PAD

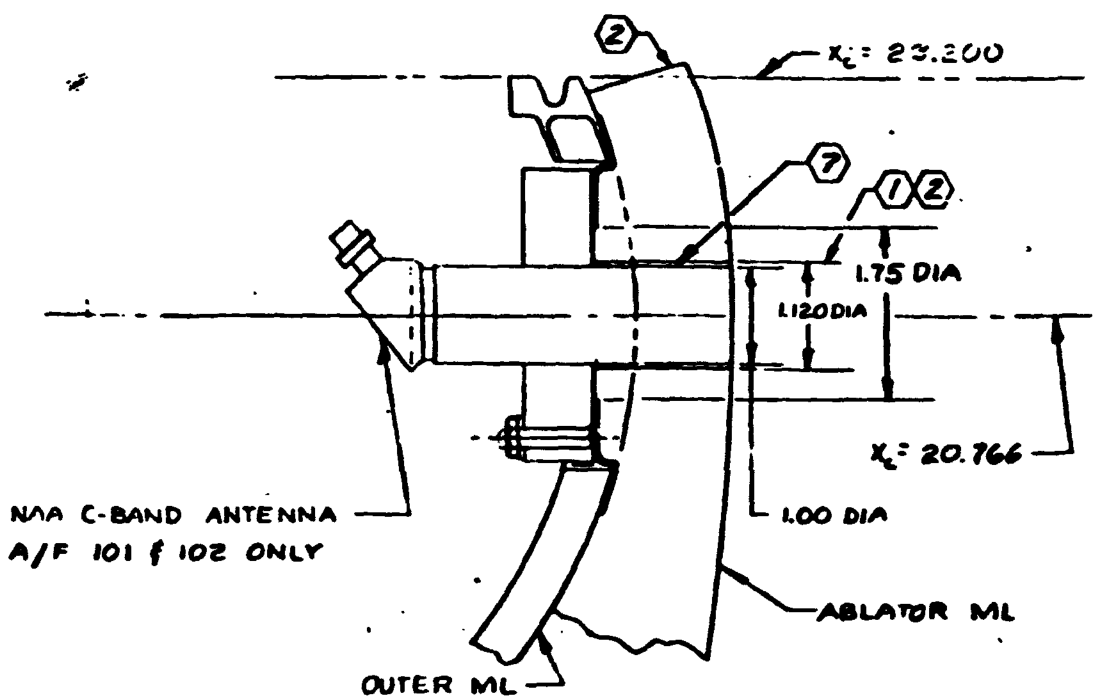


① NAA-AVCO Interface

② Nonadherent Surface.
Joint & Gasket design by AVCO.
Fabricated by NAA.

761648D

Figure 2-7 RCS FUEL DUMP



1 NAA-AVCO Interface

2 Nonadherent Surface.
Joint & Gasket design by AVCO.
Fabricated by NAA.

7 Machined by NAA.

761649D

Figure 2-8 AFT COMPARTMENT C-BAND ANTENNA (AFM 101 AND 102 ONLY)

2.4.5 Aft Compartment: S-Band Antenna (Figures 2-9)

This part is furnished, installed, and gasketed by NAA/S&ID. Note that two designs are currently effective.

No redesign of local main ablator or closeout interfaces except thickness is expected for Class B or C refurbishment. NAA/S&ID redesign of both flush and non-protruding types is expected for reuse missions to prevent aerodynamic protrusion and perturbed heating.

2.4.6 Aft Compartment: Inner Structure Attachment (Figure 2-10)

All 59 bolt plug sleeves are expected to require Class C refurbishment if current (AFM 011) practice of core-drill removal of plugs continues. No redesign of local main ablator, plug, or plug sleeves except thickness is anticipated for reuse missions.

2.4.7 Crew Compartment: Air Vents (Figure 2-11)

This part is furnished, installed, and gasketed by NAA/S&ID. Avco/SSD prepares the surrounding edgemember closeout.

No redesign of local ablator except thickness is anticipated. Reduction of local main ablator thickness will require design change to vent entrance to accept reduced ablator interface without protrusion.

2.4.8 Crew Compartment: Steam Vent (Figure 2-12)

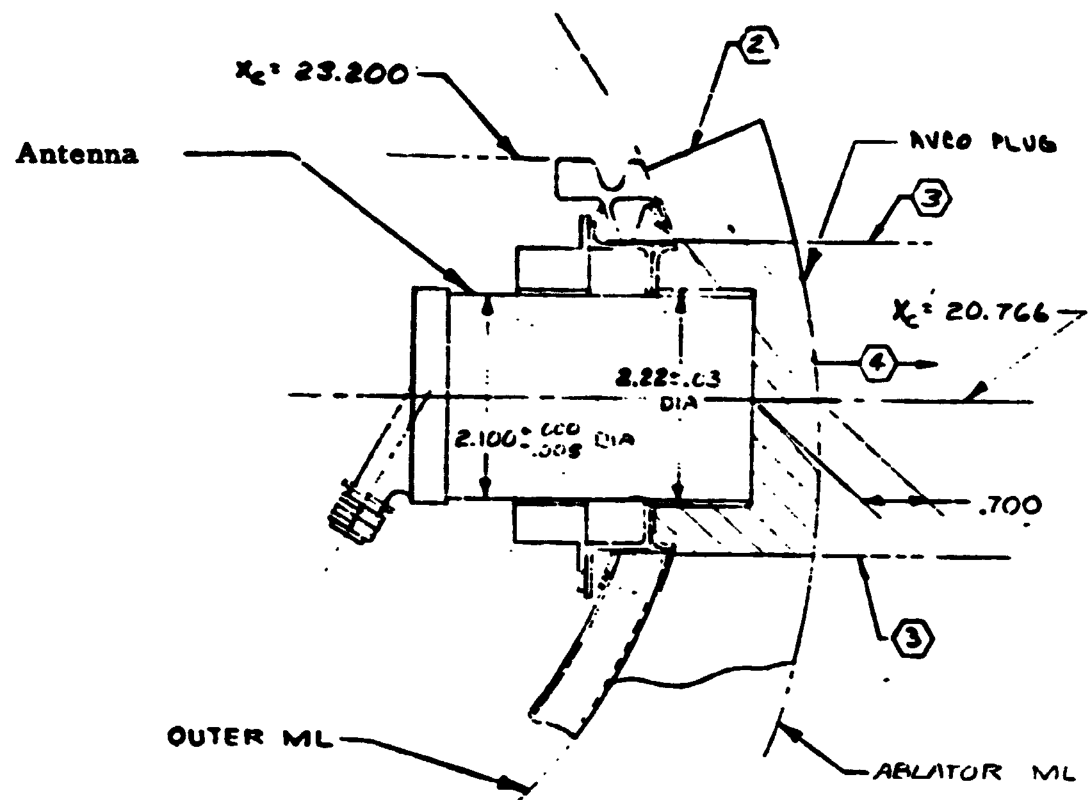
This part is furnished, installed, and gasketed by NAA/S&ID. Avco/SSD prepares the surrounding edgemember closeout.

No redesign of local main ablator except thickness is anticipated. This reduction will require design change of vent ablator interface to reduce vent entrance (molded ablator) to avoid protrusion.

2.4.9 Crew Compartment: Emergency Boost Cover Ejection Mechanism (Figure 2-13)

This part is furnished, installed, and gasketed by NAA/S&ID. Surrounding edgemember closeout is installed by Avco/SSD.

No redesign of adjacent ablator except thickness reduction is anticipated. Reduction of ablator thickness will require relocation of mechanism-cover interface unless surrounding ablator is returned to original thickness, thereby requiring Class C refurbishment. Block II mechanism redesign would not be required if the ablator plug is allowed to protrude and the resultant heating perturbation is insignificant.



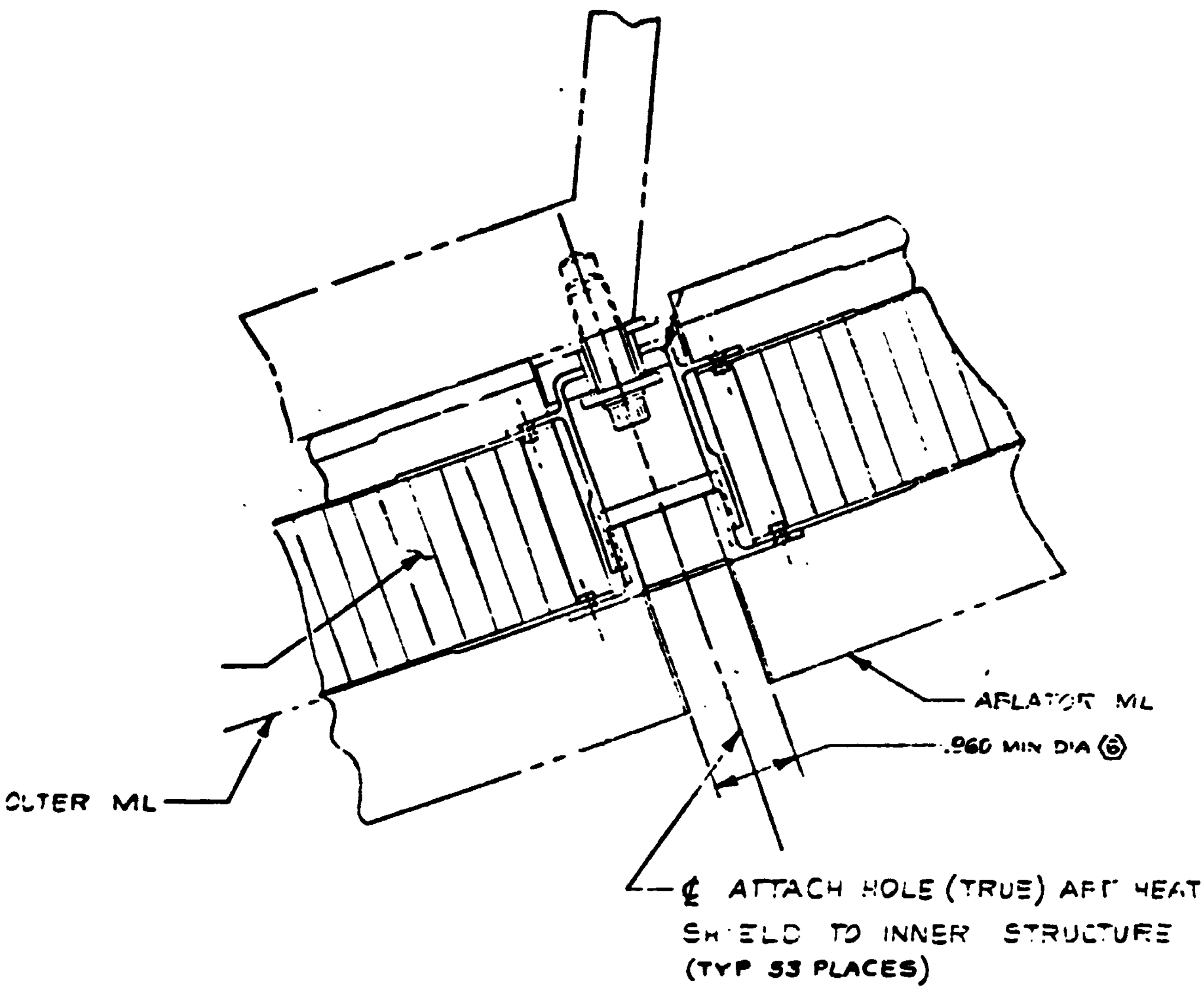
2 Nonadherent Surface. Joint & Gasket design by AVCO. Fabricated by NAA.

3 Nonadherent Surface. Joint & Gasket design by AVCO. Fabricated by AVCO.

4 Ablator design allows removal of part in direction indicated.

761650D

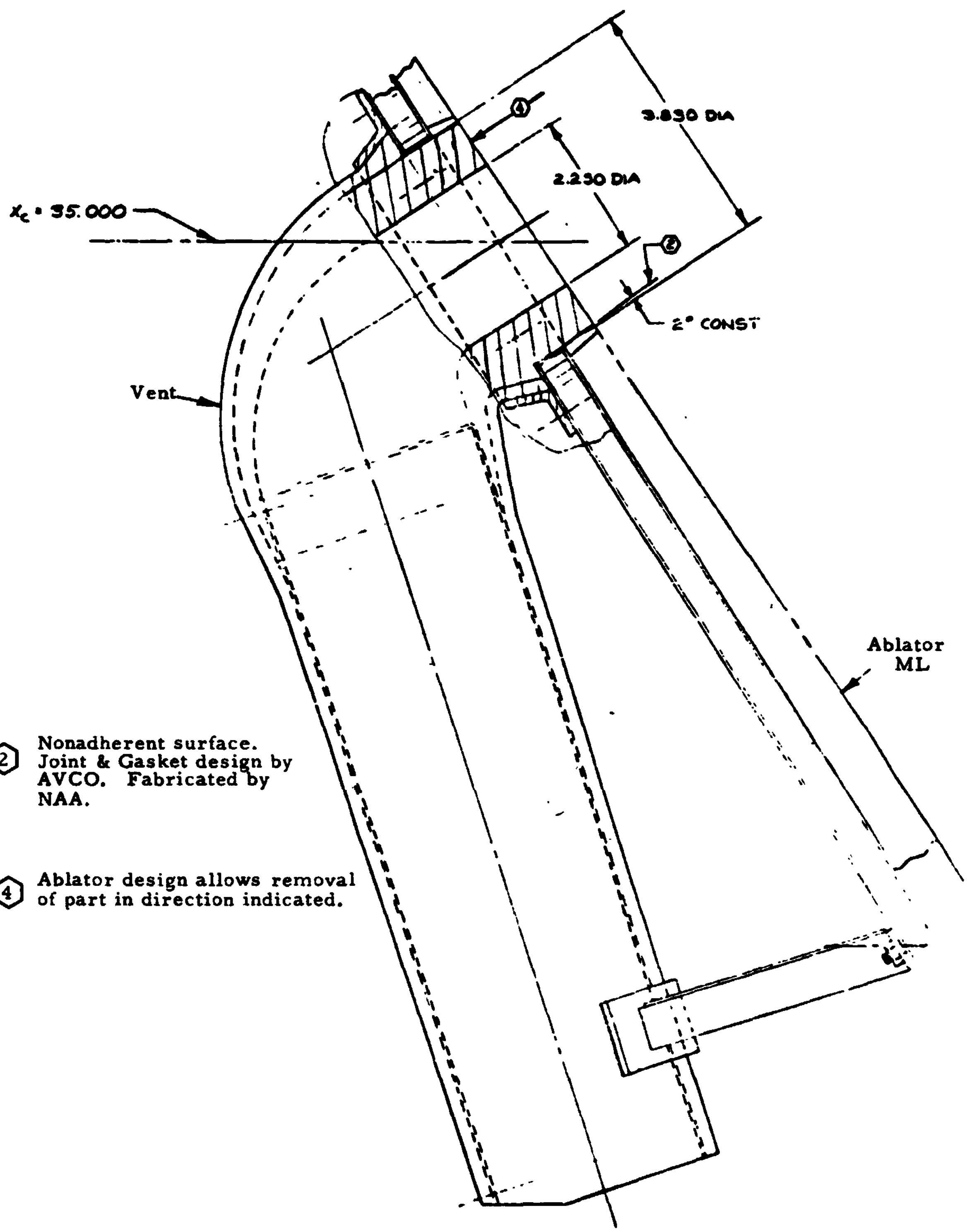
Figure 2-9 S-BAND ANTENNA, NONPROTRUDING



6 AVCO provides removable Bolt Plug with dia. shown.

761651D

Figure 2-10 AFT COMPARTMENT ATTACHMENT (TYPICAL)

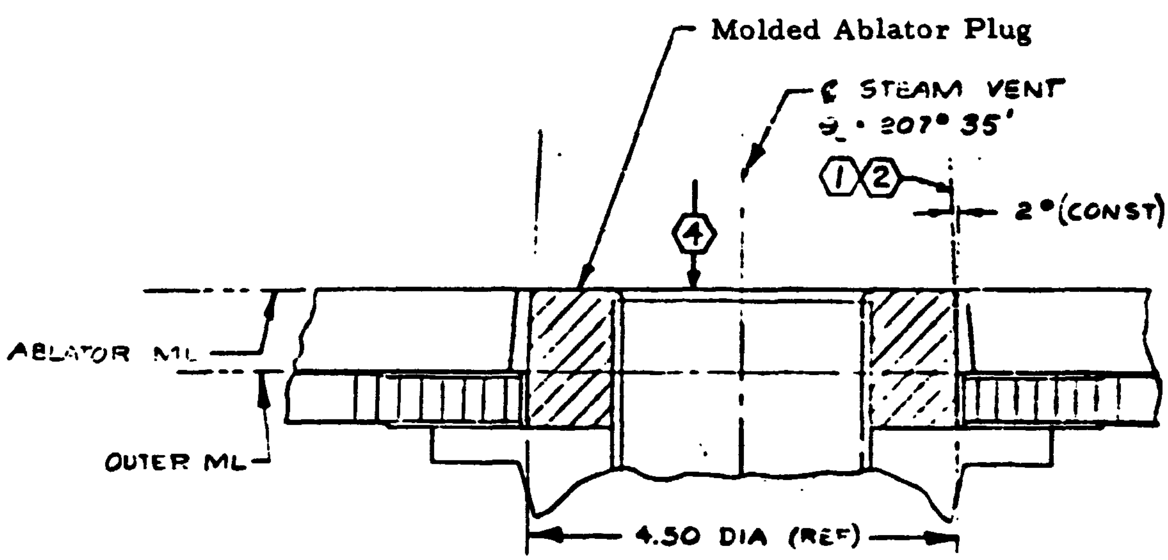


(2) Nonadherent surface. Joint & Gasket design by AVCO. Fabricated by NAA.

(4) Ablator design allows removal of part in direction indicated.

7616520

Figure 2-11 AIR VENT



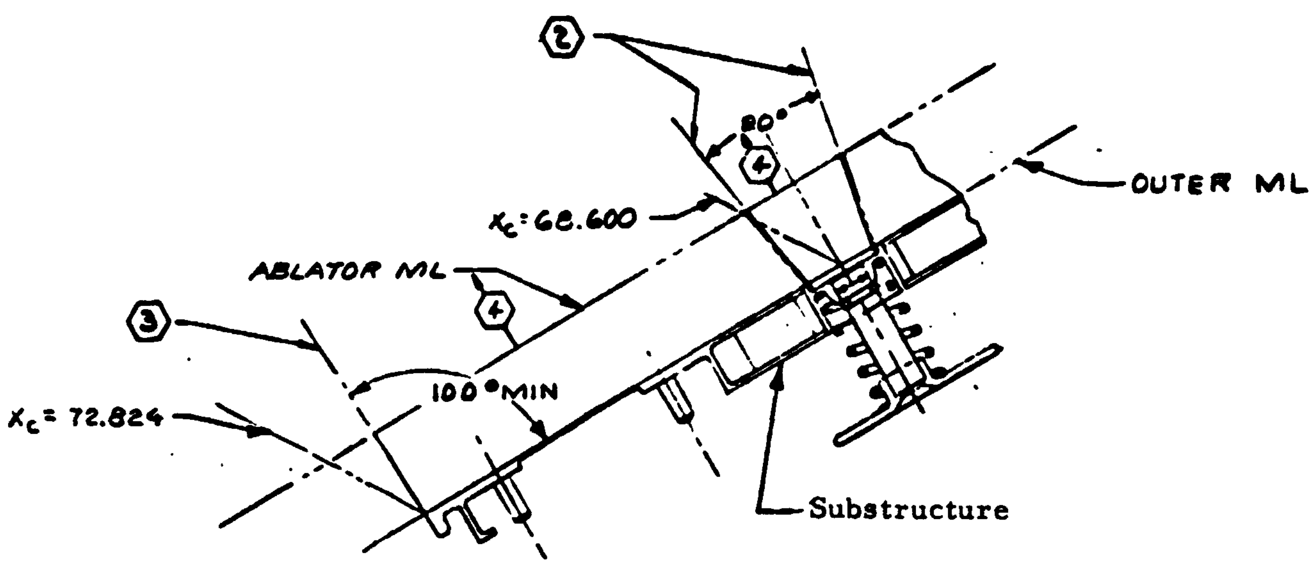
1 NAA-AVCO interface.

2 Nonadherent Surface. Joint & Gasket design by AVCO. Fabricated by NAA.

4 Ablator design allows removal of part in direction indicated.

761653D

Figure 2-12 STEAM VENT



2	Nonadherent surface. Joint & Gasket design by AVCO. Fabricated by NAA.
3	Nonadherent surface. Joint & Gasket design by AVCO. Fabricated by AVCO.
4	Ablator design allows removal of part in direction indicated.

761654D

Figure 2-13 CREW HATCH BOOST COVER EJECTION MECHANISM

2.4.10 Crew Compartment: Side Crew Hatch Access Latch (Figure 2-14)

This part is fabricated, installed, and gasketed by NAA/S&ID. Avco/SSD provides surrounding edgemember closeout.

No redesign of adjacent main ablator except thickness is anticipated. This reduction will require design change to Block II latch plug to avoid protuberance. This design change may not be required if resultant perturbed aero heating is insignificant.

2.4.11 Crew Compartment: Roll, Pitch, and Yaw Engine Panels (Figure 2-15 Typical)

Nozzle extensions for these engines are fabricated, installed and gasketed by NAA/S&ID. Avco/SSD provides surrounding edgemember closeouts.

No redesign of adjacent ablator except thickness reduction is anticipated. This reduction will require NAA/S&ID design change to associated nozzle extensions to avoid protuberance heating; otherwise Block II extensions can be used and surrounding ablator faired to original Block II thickness through Class C refurbishment.

2.4.12 Crew Compartment: Urine Dump (Figure 2-16)

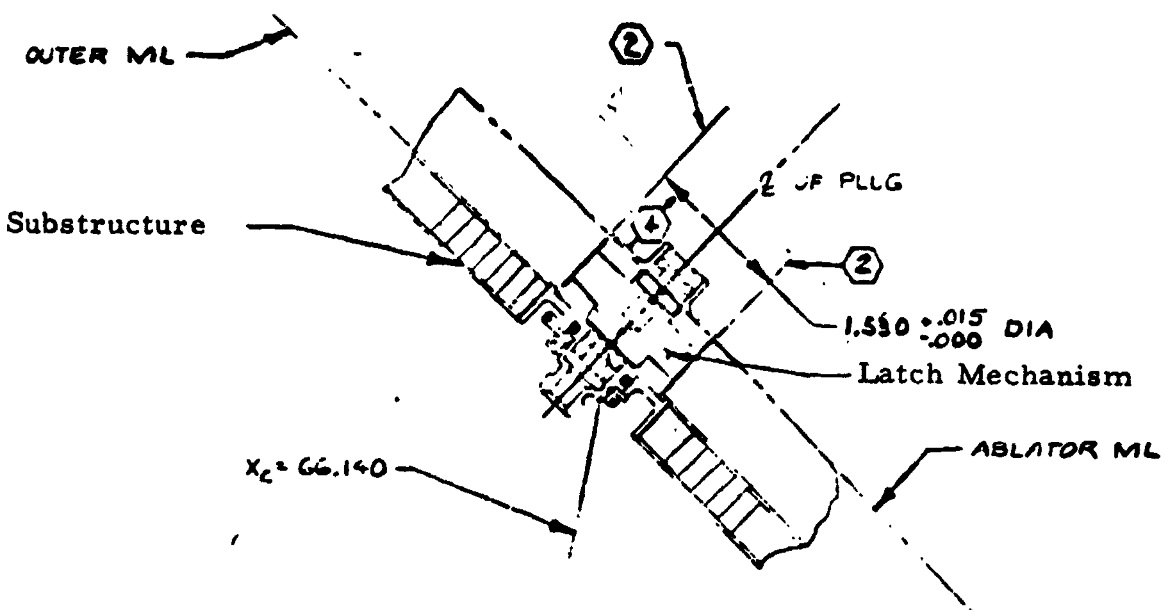
This part is fabricated, installed and gasketed by NAA/S&ID. Avco/SSD provides surrounding edgemember closeout.

No redesign of surrounding ablator except thickness is anticipated. This reduction will require design change to urine dump assembly.

2.4.13 Crew Compartment: Side Window Installation (Figure 2-17)

Side window panels are furnished, installed, and gasketed by NAA/S&ID. Avco/SSD furnishes the molded ablator panel between side window glass panels and surrounding edgemember closeouts. Molded ablator retainers for the micrometeoroid window panels are furnished, installed, and gasketed by NAA/S&ID.

Redesign would be required by a Class B refurbishment of the side window installation, i.e. to accept a decrease in thickness of surrounding ablator. Both the Avco molded panels and the NAA/S&ID molded retainers would require redesign. For Class A or Class C refurbishment of the side window installation, adjacent ablator would be faired or replaced to accept a standard Block II installation.

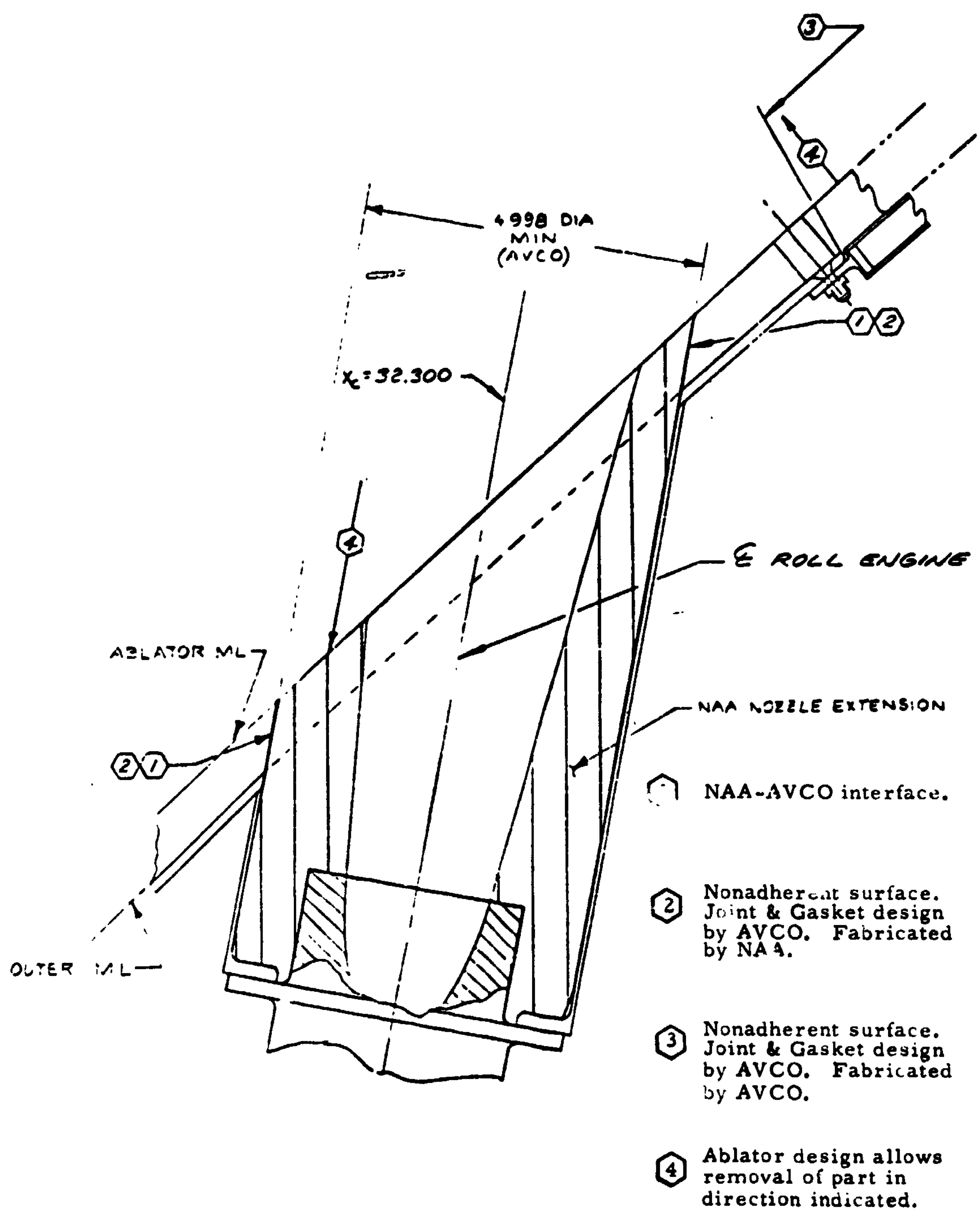


(2) Nonadherent surface. Joint & Gasket design by AVCO. Fabricated by NAA.

(4) Ablator design allows removal of part in direction indicated.

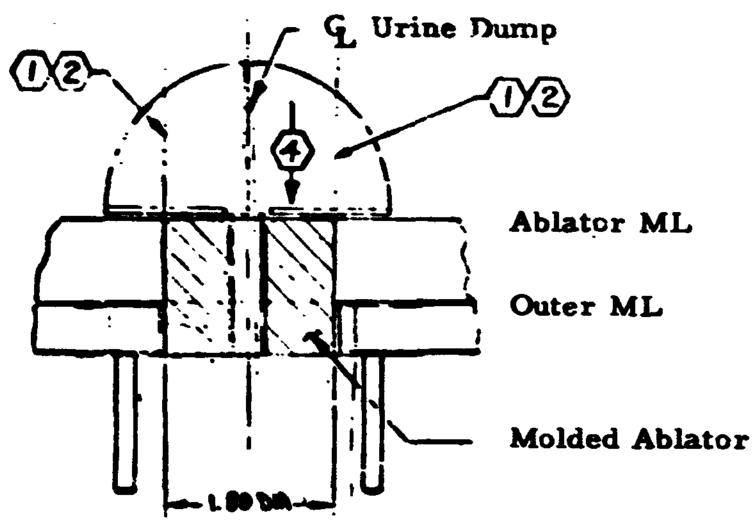
761655D

Figure 2-14 CREW HATCH LATCH MECHANISM



761656D

Figure 2-15 RCS ROLL ENGINE PANEL



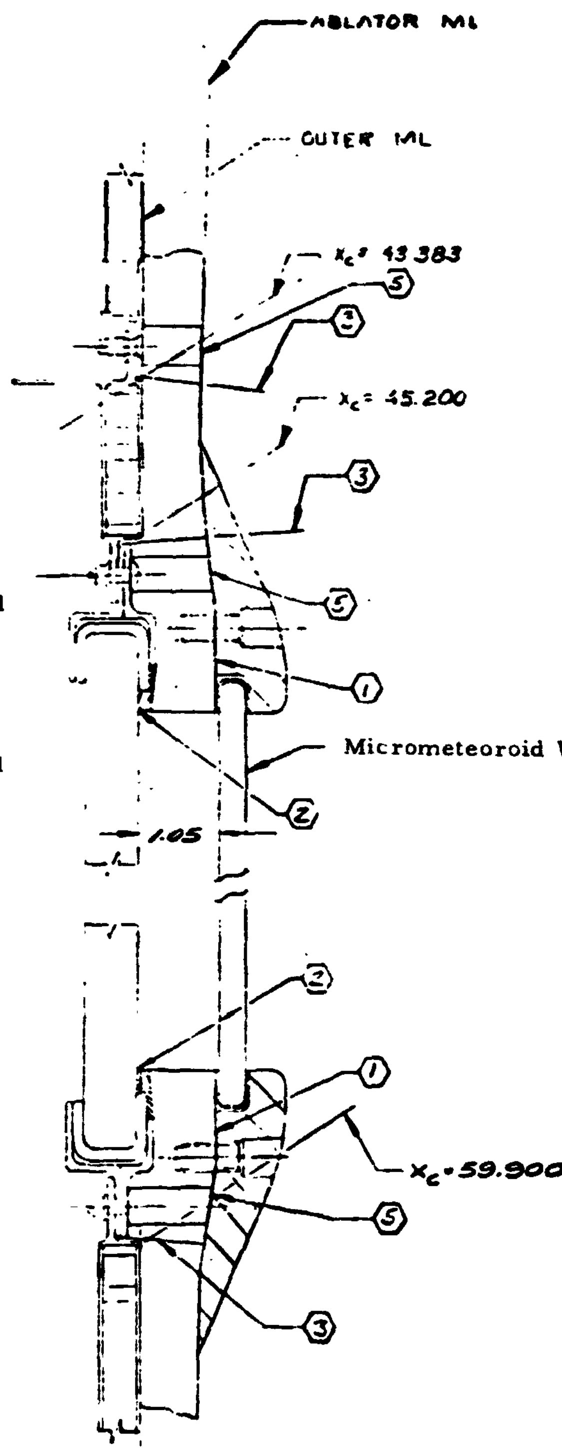
1 NAA-AVCO Interface

2 Nonadherent surface. Joint & Gasket design by AVCO. Fabricated by NAA.

4 Ablator design allows removal or part in direction indicated.

7616570

Figure 2-16 URINE DUMP



(1) NAA-AVCO interface.

(2) Nonadherent surface. Joint & Gasket design by AVCO. Fabricated by NAA.

(3) Nonadherent surface. Joint & Gasket design by AVCO. Fabricated by AVCO.

(5) AVCO to provide removable plug, min. dia. = 0.438

Figure 2-17 SIDE WINDOW INSTALLATION (TYPICAL)

2.4.14 Crew Compartment: Rendezvous Window Installation (Figure 2-18)

Rendezvous window panels are fabricated, installed, and gasketed by NAA/S&ID. Avco/SSD furnishes the molded ablator window frames and well panel.

For Class C refurbishment minor design changes to frames and to the well panel would be required to accept a decrease in thickness of the surrounding crew compartment main ablator or even to accommodate a decrease in thickness within the well. No redesign would be required by a Class A refurbishment or by a Class C refurbishment wherein the crew compartment ablator is faired into a standard Block II installation. The latter approach would, of course, require a Class C refurbishment of the surrounding crew compartment ablator as well as the window installation.

2.4.15 Crew Compartment: C-Band Antenna (AFM 101 & 102 only) (Figure 2-19)

This part is fabricated, installed, and gasketed by NAA/S&ID. Avco/SSD provides surrounding closeout and gaskets antenna panel.

This installation is applicable to AFM 101 and 102 only and does not appear on subsequent vehicles. It is anticipated that this installation will be removed for any reuse mission through design change and that a local Class C refurbishment or a replacement plug will be provided to accommodate this change. Class A, B, or C refurbishment would require no redesign of local ablator except thickness if this installation were retained for a reuse mission.

2.4.16 Forward Crew Hatch (Figure 2-20)

This part will be recovered. No difficulty is foreseen on Class A, B, or C refurbishment. New design thicknesses would be required for Class B refurbishment. In local regions such as that for the latch operation mechanism, Class C refurbishment or redesign may be required.

2.4.17 Hatch or Access Panel Frames (Figure 2-21, Typical)

These frames are provided by Avco/SSD for general hatch or panel access and installation. No design change to these frames except thickness reduction is anticipated for Class A, B, or C refurbishment.

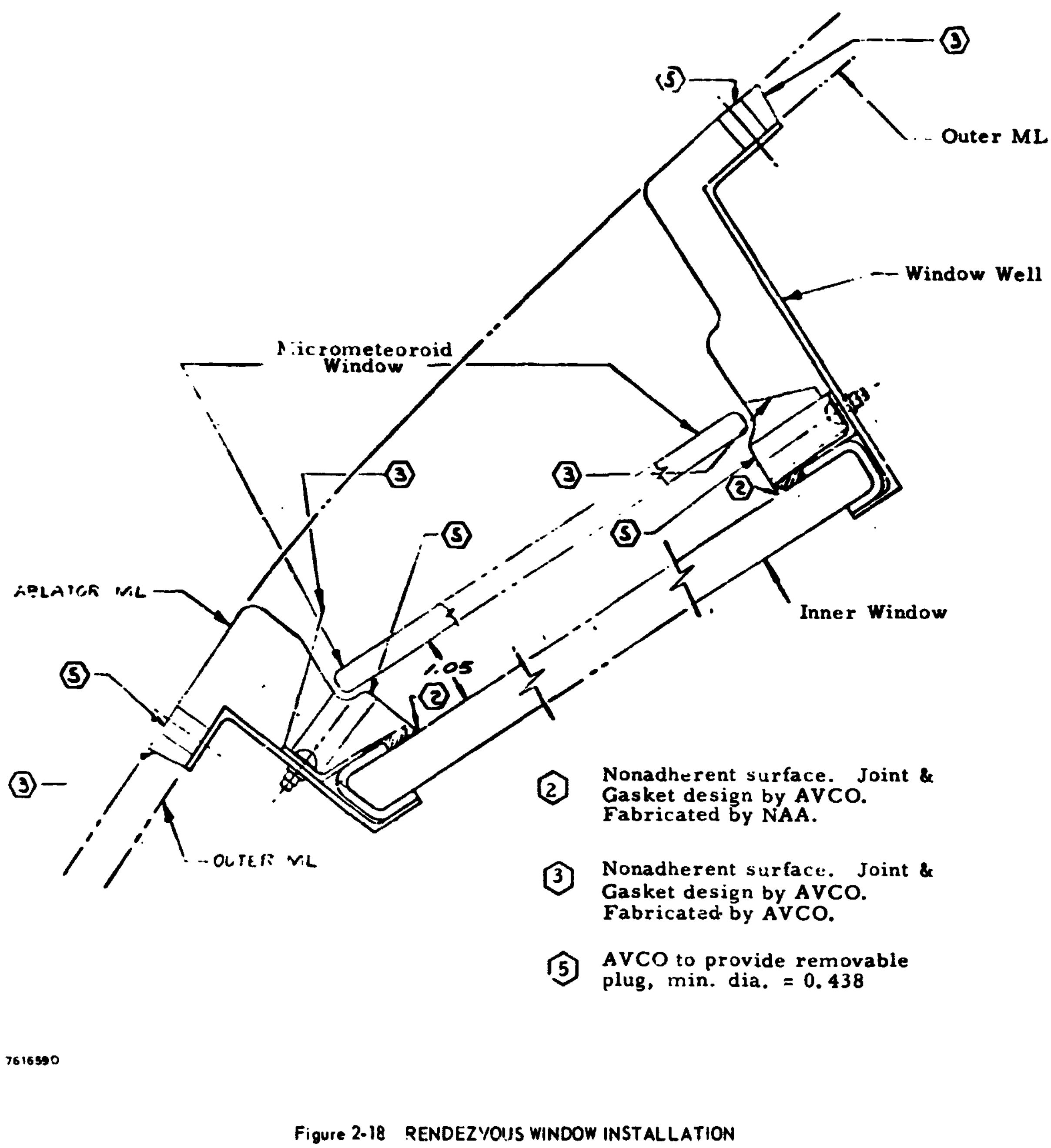
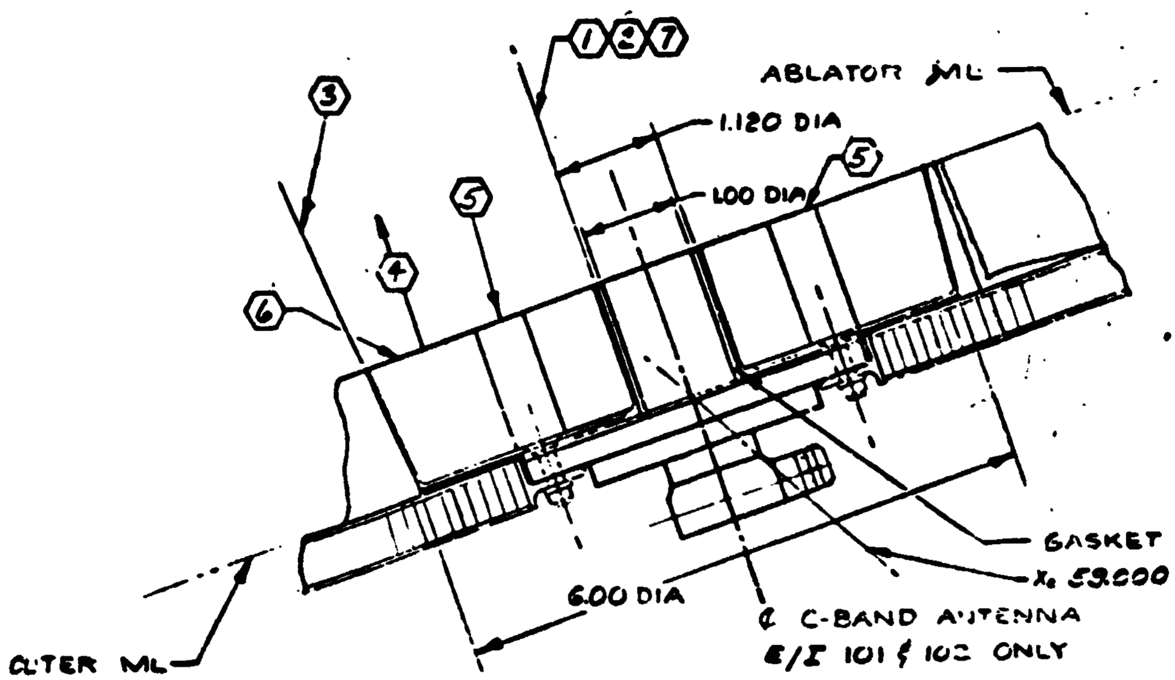


Figure 2-18 RENDEZVOUS WINDOW INSTALLATION



1. NAA-AVCO interface.

2. Nonadherent surface. Joint & Gasket design by AVCO. Fabricated by NAA.

3. Nonadherent surface. Joint & Gasket design by AVCO. Fabricated by AVCO.

4. Ablator design allows removal of part in direction indicated.

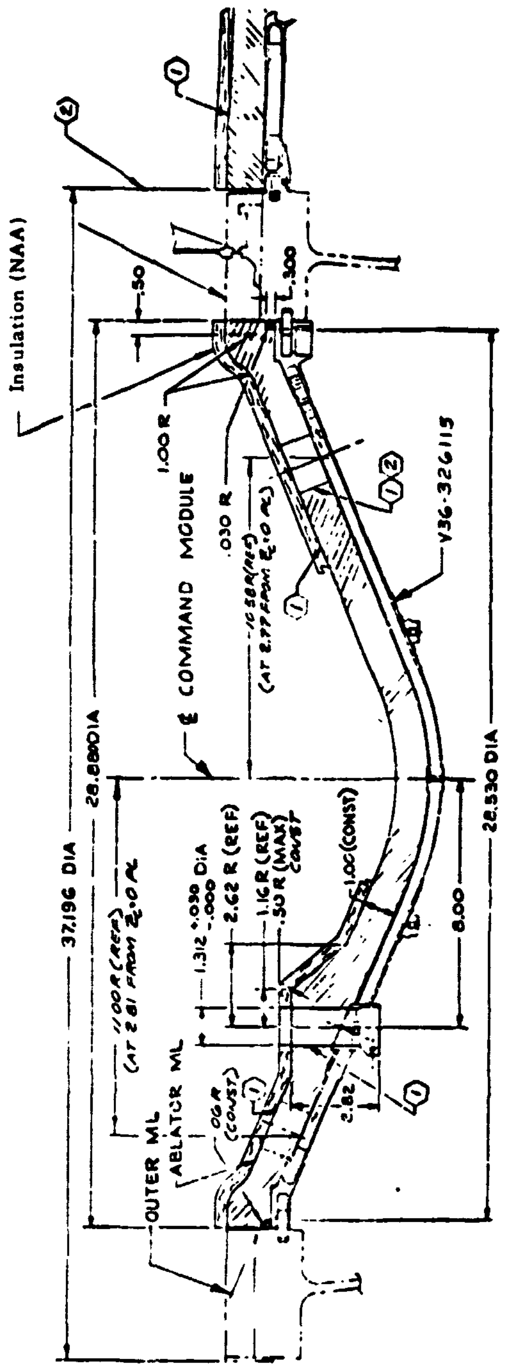
5. AVCO to provide removable plug, min. dia. = 0.438

6. AVCO provides removable plug with dia. shown.

7. Machined by NAA.

761660D

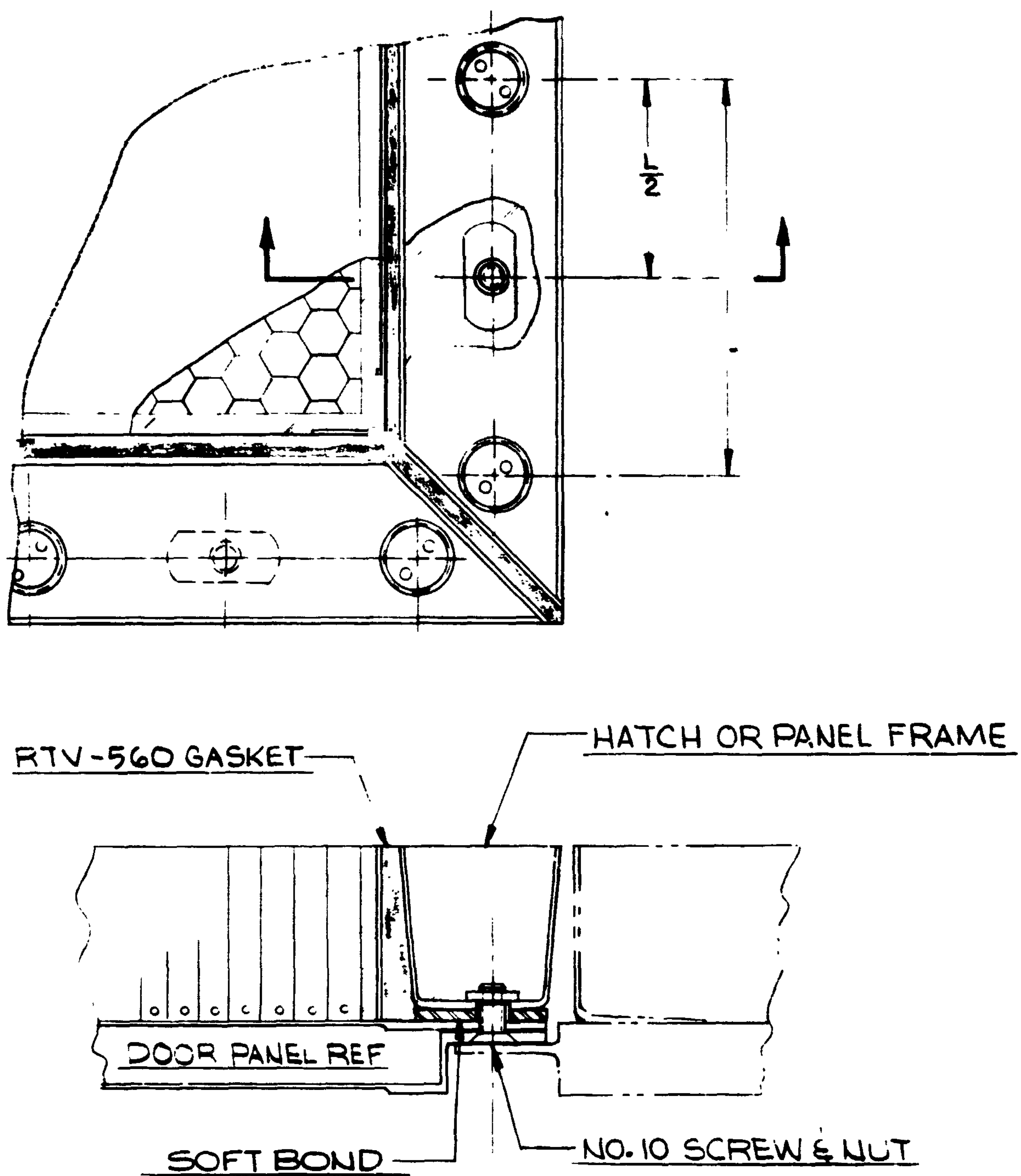
Figure 2-19 CREW COMPARTMENT C-BAND ANTENNA (AFM 101 AND 102 ONLY)



① NAA-AVCO interface.

② Nonadherent surface. Joint & Gasket design by AVCO. Fabricated by NAA.

Figure 2-20 FORWARD CREW HATCH



7616620

Figure 2-21 HATCH AND ACCESS PANEL FRAME (TYPICAL)

No redesign of gaskets surrounding crew compartment or forward compartment hatches or inspection panels appears to be required by Class A, B, or C refurbishment.

2.4.18 Intercompartment Interfaces and Gaskets

The Block II forward - crew compartment interface and gasket are provided by Avco/SSD. The Block II crew-aft compartment interface and gasket are provided by NAA/SSD. Class A refurbishment will not require design change to these interfaces or gaskets. Class B or C refurbishment will require design change for reduced thickness only.

In general, Class C refurbishment of the aft compartment at the crew compartment interface, or local fairing of the crew compartment, will be required for a Class A refurbishment of the crew compartment in order to provide a smooth interface.

2.5 STRUCTURAL DISTORTION

It is possible that under certain landing conditions the stainless steel substructure, particularly in the aft compartment, may sustain local permanent distortion. Such distortion will require NAA/S&ID inspection, NDT, and repair prior to shipment of the recovered heat shield to Avco/SSD for refurbishment. This will require removal at NAA/S&ID of local uncharred ablator from distorted regions. Steel repair will require procedures that do not deteriorate adjacent ablator unless Class C refurbishment of this ablator is anticipated. In general, the feasibility of Class C refurbishment appears unaffected by structural distortion. The feasibility of Class A or B refurbishment will depend upon the ability to determine nondestructively the presence and extent of any bond delamination that might occur as a result of water impact or structural distortion. At this time, it appears that ultrasonic NDT techniques may be used to detect such delamination, but verification of this presumption is required.

Structural distortion or water impact may also cause local honeycomb node separation which would require local Class C refurbishment or repair. X-Ray techniques are expected to be adequate to assess such damage.

In summary, before the feasibility of Class A and B refurbishment can be demonstrated, it is necessary to assess the ability to detect bond delamination in retained ablator which has undergone water impact or other structural or thermostructural loading.

Structural distortion or bond delamination have no effect on the feasibility of Class C refurbishment.

2.6 MANUFACTURING AND QUALITY CONTROL

Refurbishment of the Block II Apollo Command Module heat shield will require the following major manufacturing and QC activities:

a) Ablator Inspection

b) Ablator Removal

c) Ablator Refurbishment and Finishing

It can be stated at this time that, from the standpoint of manufacturing, Class A, B, and C refurbishments of the Apollo heat shield appear feasible. As noted earlier, Class C refurbishment is essentially no different than a Class 3 repair which is currently performed regularly on Block II hardware. Furthermore, no manufacturing process has been identified in any of these refurbishments which requires skills or equipment not already developed and in use at Avco/SSD on the Block II Apollo program.

Class C refurbishment will require no development or expansion of QC skills or equipment. Class A and Class B refurbishments, however, will require some limited expansion of QC technology. As noted in Sections 2.1 and 2.3, development and verification of NDT and QC production techniques and tools for determining residual char and salt water content will be required. No other QC skills or equipment have been identified which are not already in use on the Block II program.

2.6.1 Ablator Inspection

A key activity in the refurbishment of the Block II heat shield ablator will be the determination of the degraded state of the recovered heat shield. This information is required, in general, for each ablator area and heat shield component before the class of refurbishment for that area or component can be selected. Included in these inspection data are the following:

a) damage assessment: data on local cracking, ablator delamination, and edgemember delamination

b) char depth: depth of non-reusable ablator

c) major dimensional data: data for manufacturing process and machining tape generations

d) water content: data on ablator water content for determination of drying requirements

In addition to these specific inspection data, standard Block II process and fabrication QC data will be required.

An important question should be raised here as to the order in which the above inspection data should be obtained. Specifically, should char depth damage and water content data be obtained and the local class of refurbishment selected prior to vehicle machining or should machining to final contour be accomplished first? (In either case major dimensional data are the first order of business.) At this stage in the investigation it appears that it would be more efficient and economical to use the latter approach. With the ablator machined to final contour for the reuse mission more reliable NDT and QC data can be obtained on damage, char remaining, and the water content of reusable ablator. In addition, it appears that these data could be obtained with a minimum of complexity and cost. It appears that a more reliable selection of the class of refurbishment could then be made.

Any approach to refurbishment is expected to require the drilling of thickness inspection holes in the ablator, a standard procedure currently employed in the Block II manufacturing process. These holes would be located and drilled following receiving inspection to obtain major dimensional data for machining and to obtain subsequent manufacturing and QC data.

2.6.2 Ablator Removal

As noted earlier, no manufacturing process has been identified, including the removal of charred or degraded ablator for Class B refurbishments which requires skills or equipment not already developed and in use on the Block II Apollo program. The same tape-controlled machining techniques and equipment would be utilized in charred ablator removal. For the approach wherein the charred ablator is removed to the final ablator thickness for reuse, machining techniques are expected to be identical with current Apollo machining procedures. Similarly, existing Apollo techniques and equipment will be used for final contouring and finishing of Class C refurbishments.

Class A refurbishment is essentially a rejuvenation of ablator surfaces which have undergone relatively light or incomplete charring. Should Class A areas require removal of thin surface layers of degraded materials such as paint, it is anticipated that careful hand finishing techniques (currently used extensively on Block II Apollo) will be utilized.

2.6.3 Ablator Refurbishment and Finishing

In Class A and Class B refurbishments of main ablator only two manufacturing refurbishment or rejunevation processes are anticipated following removal of surface degradation. These are application of the epoxy pore sealer and application of the white epoxy paint. Both of these would be carried out in accordance with existing Apollo specifications.

As noted earlier, for Class C refurbishments, i.e., total replacement of local ablator with new ablator, Apollo Class 3 repair procedures would be followed. In this process the used or defective ablator material is completely removed down to the ablator-HT 424 adhesive interface. This removal is generally accomplished rapidly with special hand operated tools designed to prevent penetration of the HT 424 and damage to the substructure. The interface with adjacent, reusable ablator is maintained square or vertical to the local substructure. The base HT 424 primer is then applied in accordance with Apollo specifications. A new layer of HT 424 tape adhesive is applied and the open ablator honeycomb bonded and cured. A gap splice is maintained around the periphery of the repair or refurbishment. The open honeycomb is then primed and the ablative material gunned into the cells in accordance with standard Apollo specifications. The repaired or refurbished area is then cured at 200 - 210°F and the surface finished to contour in accordance with local requirements. Larger areas are generally tape machined and small areas are generally hand-finished. The resulting repair or refurbished ablator is then pore sealed and painted in accordance with Apollo process specifications.

Edgemember refurbishment can be carried out without total replacement of the edgemember, i.e., if damage of the edgemember is nominal. Post-flight examination of AFM 011 edgemembers indicated excellent performance without evidence of exaggerated charring or of detectable delamination. For Class A and B refurbishments, no processing of edgemembers is anticipated except a reduction in local height to match ablator thickness requirements. Local delaminations or damage may be repaired by resin impregnation and/or doubling, depending on damage assessment,with no effect upon performance. Such repairs or refurbishments have been used and flight tested on AFM 009 and AFM 011.

Gasket refurbishment can also be carried out without total replacement of the the gasket if it has suffered no substantial degradation or damage from flight. Examination of AFM 011 has also indicated excellent performance for the RTV-560 gaskets. With one exception all crew compartment and intercompartmental gaskets showed no differential recession. Based upon the results of AFM 011 it appears for Class B refurbishment that many crew compartment gaskets would require no refurbishment save a reduction in height to reuse dimension. Class C refurbishment of surrounding ablator would, of course, require Class C refurbishment of the gasket. It is not known whether Class C gasket refurbishment will be required in areas of Class A ablator refurbishment. This will be entirely dependent on local gasket performance, on the relative depth of swelling, and on handling care.

Table 2-4 summarizes the types of refurbishment currently anticipated for some of the major heat shield areas and components. Alternate combinations are presented since the decision between A, B, or C refurbishment of local ablator and components is a strong function of local char thickness and individual vehicle performance with the performance at lunar velocities still unknown.

TABLE 2-4

ANTICIPATED REFURBISHMENT REQUIREMENTS FOR MAJOR APOLLO HEAT SHIELD AREAS OR COMPONENTS

Heat Shield Area or Component	Class of Refurbishment			
	Main Ablator	Edgemembers	Gaskets	Other
Aft Compartment				
Shear and compression pads	B	B	C	Pads - C
	C	C	C	Pads - C
RCS oxidizer and fuel dumps	B	B	B, C	*plugs - C
	C	C	C	*plugs - C
C-Band antenna (101 and 102 only)	B	B	C	*antenna - removal, plug - C
S-Band antenna	C	Removal	Removal	*antenna - removal
	B	B	B, C	
	C	C	C	
59 bolt plug areas	C	--	--	Sleeves - C, plugs - C
Crew Compartment				
Air vents	A	A	A, C	
	B	B	B, C	
	C	C	C	
Steam vent	A	A	A, C	
	B	B	B, C	
	C	C	C	
Boost cover ejection mechanism	A	A	A, C	
	B	B	B, C	
	C	C	C	
Crew hatch access latch	A	A	A, C	
	B	B	B, C	
	C	C	C	
Roll, pitch, and yaw engine panels	B	B	C	
	C	C	C	
Urine dump	B	B	B, C	
	C	C	C	
Side and rendezvous window installations	A	A	A, C	Frames - A, *Retainers - A
	B	B	C	Frames - C, *Retainers - C
	C	C	C	Frames - C, *Retainers - C
C-Band antenna (101 and 102 only)	A	A	C	*Antenna - removal, *plug - C
	B	B	C	*Antenna - removal, *plug - C
	C	Removal	Removal	*Antenna - removal
Forward Compartment	C	C	C	(new compartment)
Other				
Forward crew hatch	A	A	A, C	*Insulation - C
	B	B	B, C	Insulation - C
Hatch or Inspection Panel access frames	A	A	A, C	Sleeves - A, plugs - C
	B	B	B, C	Sleeves - B, C, plugs - C
	C	C	C	Sleeves - C, plugs - C
Intercompartment interfaces and gaskets	A	A	C	
	B	B	C	
	C	C	C	

A Class A (total reuse)
B Class B (reduce thickness and reuse)
C Class C (totally replace with new)

*NAA/S&ID Furnished

The following additional notations should be made with respect to areas and components listed on Table 2-4.

2.6.4 Aft Compartment: Shear and Compression Pads (Figures 2-5 and 2-6)

Particular difficulty is anticipated in removing shear and compression pads, since these components are of relatively high density and are bonded to the substructure. Removal by grinding appears required. This difficulty is compounded by the presence of the interrupted steel retaining ring at the base of each pad. This ring is bonded to the substructure and will require extremely delicate removal with tools which can potentially damage the substructure. It is highly probable that the removal of these pads will necessitate damage to adjacent edgemembers and adjacent ablator, thereby requiring a Class C refurbishment of these elements. It is also possible that shear pad removal will damage the Marinite installation in the substructure, thereby requiring a Class C refurbishment of the Marinite.

2.6.5 C-Band Antenna Installations

Removal of these antennas from AFM 101 aft compartment is anticipated. With an alteration of local substructure class C refurbishment of the local ablator would remove edgemembers and gaskets. Class A and B refurbishments could retain or remove edgemembers, depending on NAA/S&ID modifications, in which case the antenna would be replaced with a plug or new ablator.

2.6.6 Window and Panel Access Frame Bolt Plug Installations

Refurbishment of local bolt plug installations, sleeves and adjacent ablator, is a strong function of the difficulty experienced in removing each bolt plug. If bolt plugs can be removed without damage to the sleeves, then Class A and B refurbishments of the sleeves is feasible. Should sleeves be damaged in disassembly, repair techniques are available and proven for replacing the sleeves, including substitution of cast sleeves or a Class C refurbishment. Class C refurbishment of bolt plug sleeves would entail Class C refurbishment of adjacent ablator in panel frames and complete replacement of window frames. Damage to panel frame edgemembers during removal will generally require complete replacement with a new panel frame.

2.7 IMPACT OF AFM 017 AND AFM 020 FLIGHT TEST DATA

The evaluation of Apollo heat shield refurbishment requirements summarized in this report is based upon an understanding and projection of heat shield performance derived from AFM 009 and AFM 011 post-flight evaluations. The impact of post-flight evaluation of AFM 017 and AFM 020 on Class A and Class B refurbishment is currently unknown. It is expected that the greater heating on these flights will result in heat shield response approximating that

of entry from lunar return. The major contribution of AFM 017 and AFM 020 to assessing feasibility of refurbishability will be the generation of operational char thicknesses, local recession and aggravation, and the acquisition of thermal performance data (temperatures, flux rates, etc.). These data and observations will permit the selection of local classes of refurbishment with much greater certainty. The certainty of use of Class A and Class B refurbishments should be sufficiently clarified to permit preparation of detailed refurbishment plans and to permit generation of realistic cost estimates.

Class C refurbishment is currently considered feasible; AFM 017 and AFM 020 data are not expected to have any impact on the feasibility of Class C refurbishment.

The acquisition of the following post-flight test data from AFM's 017 and 020 is expected to significantly enhance the assessment of feasibility of Class A and B refurbishments.

2.7.1 Density-Temperature Gradients

Post-flight core samples should be taken from typical regions of the vehicle in the near vicinity of temperature sensors from which good flight test data have been recovered. These core samples should be used to obtain char density and NDT char depth data. These data, in combination with the temperature data, will aid significantly in assessing the feasibility of char depth measurement, refurbishment class selection, and reused ablator thermal history and performance.

2.7.2 Water Content Measurements

Post-flight cores should be taken from selected regions for direct quantitative measurement of salt water retained from recovery of actual flight vehicles and for calibration of potential NDT techniques and tests for measurement of water content.

2.7.3 Char-depth Distribution

Post-flight cores should be taken from as many vehicle locations as practical (including the aforementioned cores and anticipated local problem areas) in order to obtain char-virgin thickness distributions of sufficient detail to permit reliable definition of potential Class A and Class B refurbishment areas. These analyses should be complemented with exhaustive post-flight examination of degradation of local components such as bolt plugs, bolt plug sleeves, edge-members, frames, gaskets, pads, etc.

3.0 CONCEPTUAL REFURBISHMENT PLAN

Although it is not possible at this time to prepare a true, definitive plan for the refurbishment of Block II heat shields for earth orbit reuse, the presentation of a conceptual refurbishment plan helps to identify the underlying logic of refurbishment by identifying expected major tasks and their interrelations. The conceptual refurbishment plan outlined here does not include those engineering, development, and test tasks which are associated with demonstrating feasibility. It is assumed that such tasks demonstrating feasibility have been executed prior to implementation of any refurbishment plan.

This conceptual refurbishment plan is based on the following assumptions, guidelines, and constraints. In a number of cases individual guidelines are best assumptions based upon available information; the validity of the resulting plan is largely dependent on the validity of these assumptions.

a) All heat shield panels and components for all vehicles will be refurbished for reuse on low earth orbit missions of 14 days duration.

b) Design criteria for refurbished heat shields will be identical with Block II Apollo criteria.

c) NAA/S&ID will perform major vehicle disassembly, including compartment disassembly and removal of NAA furnished components. NAA/S&ID will also perform all substructure cleaning and repair prior to shipment to Avco/SSD.

d) Avco/SSD will perform refurbishment of all heat shield panels and components for which it has Block II responsibility, including,

1) removal of all degraded or non-reusable ablator and components

2) Class A, B, or C refurbishment and finishing.

e) Heat shields of five Block II vehicles recovered from earth orbital and lunar missions will be refurbished:

1) AFM 101: Earth orbital

2) AFM 103: Lunar

3) AFM 105: Earth orbital

4) AFM 107: Lunar

5) AFM 108: Lunar

f) Maximum utilization will be made of Block II Apollo, materials, processes, specifications, and tooling.

g) A minimum turnaround time of one year from Block II recovery to second launch is required. NAA/S&ID will require delivery of refurbished heat shields a minimum of 5 months prior to launch.

h) Class A and Class B refurbishments will, in practice, be proven feasible for earth orbital and lunar heat shields and for aft and crew compartment areas and components. In particular, acceptable performance and/or revised design criteria for ablator reuse will be demonstrated.

3.1 BASELINE REFURBISHMENT

Because of the complexity and interdependence of such factors as Block II entry trajectory, local ablator and component performance, water impact and recovery environment, char depth definition, cost, and schedules, all of the potential combinations of approaches cannot be presented in this plan. Instead, a single, conceptual, baseline approach to refurbishment is proposed which separately provides for the refurbishment of Block II earth orbital and lunar vehicles. Total Class C refurbishment is discussed in addition to the baseline for comparison purposes.

Table 3-1 and 3-2 summarize the baseline refurbishments conceived for earth orbital and lunar heat shields, respectively. The earth orbital baseline anticipates Class B refurbishment of aft compartment and crew compartment main ablator. Some local class C refurbishment of the roll engine panel ablator and edge members is also anticipated. Class A refurbishment of forward crew hatch ablator is also anticipated under the assumption that there are no interface effects. Class A refurbishment of window installations is also anticipated with replacement of NAA window panel gaskets expected. In general, class C refurbishment of almost all gaskets is anticipated. Some access panel frames may not require new gasketing; but, in general, reduction of ablator thickness in Class B refurbishment for the reuse mission is expected to alter local panel distortions sufficiently to make regasketing preferable. It is also expected that certain bolt plug sleeves will require refurbishment because of damage on removal of plugs.

The lunar baseline described in Table 3-2 anticipates Class C refurbishment of the entire aft compartment. Class B refurbishment of the crew compartment main ablator is expected with the exception of certain local areas. Class C refurbishment of the window installations is anticipated in order to avoid redesign (see 2.4).

TABLE 3-1

BASELINE REFURBISHMENT PLAN: EARTH ORBIT HEAT SHIELDS

Heat Shield Area or Component	Class of Refurbishment			
	Main Ablator	Edgemembers	Gaskets	Others
Aft Compartment				
Main Ablator-all quadrants	B	-	-	
Shear & Compression pads	B	B	C	Pads - C
RCS Oxidizer and Fuel Dumps	B	B	C	*Plugs - C
C-Band Antenna (101 only)	B	B	C	*Antenna-remove; replace with plug
S-Band Antenna	B	B	C	
59 bolt Plug Areas	B	-	-	Sleeves-C; Plugs-C
Crew Compartment				
Main Ablator-all quadrants	B	-	-	
Air Vents	B	B	C	
Steam Vent	B	B	C	
Boost Cover Ejection Mechanism	B	B	C	*Plug-C
Crew Hatch access Latch	B	B	C	*Mechanism-C
Roll, Pitch, and Yaw Engine Panels	B, C	B, C	C	
Urine Dump	B	B	C	
Side & Rendezvous Window Installations	A	A	A, C	Frames - A; *Retainers - A
C-Band Antenna (101 only)	B	B	C	*Antenna remove; replace with plug
Forward Compartment	C	C	C	(new compartment)
Other				
Forward Crew Hatch	A	A	C	*Insulation - C
Hatch or Access Panel Frames	B	B	B, C	Sleeves - B, C; Plugs - C
Intercompartmental Interfaces & Gaskets	B	B	C	

A: Class A (total reu:
B: Class B (reduce thi. ness & reuse)
C: Clars C (totally replace with new)

*NAA/S&ID furnished

TABLE 3-2

BASELINE REFURBISHMENT PLAN: LUNAR HEAT SHIELDS

Heat Shield Area on Component	Class of Refurbishment			
	Main Ablator	Edgemembers	Gaskets	Others
Aft Compartment	C	C	C	C
Crew Compartment				
Main Ablator - all quadrants	B	-	-	
Air Vents	B	B	C	
Steam Vent	B	B	C	
Boost Cover Ejection Mechanism	B	B	C	*Plug - C
Crew Hatch Access Latch	B	B	C	*Mechanism - C
Roll, Pitch, and Yaw Engine Panels	B, C	B, C	C	
Urine Dump	B	B	C	
Side & Rendezvous window Installations	C	C	C	Frames - C *Retainers - C
Forward Compartment Other	C	C	C	(new compartment)
Forward Crew Hatch	B	B	C	*Insulation - C
Hatch of Access Panel Frames	B	B, C	B, C	Sleeves - B, C: Plugs - C
Intercompartmental Gaskets	-	-	C	

A: Class A (total reuse)
B: Class B (reduce thickness & reuse)
C: Class C (totally replace with new)

*NAA/S&ID furnished.

Figures 3-1 and 3-2 give typical examples of baseline refurbishment of earth orbital and lunar hardware. It must be understood that the refurbishments described in Figures 3-1 and 3-2 are typical examples only and that individual vehicle problems can be expected to induce minor changes to processing logic and detail. These baseline refurbishments are founded on the concept of identification of char depth and selection of final local refurbishment class after machining the ablator to reuse thickness. This approach is currently considered less costly and time consuming than that of determining char depth and refurbishment approach prior to machining.

Since the baseline refurbishment assumed here anticipates Class B refurbishment of both earth orbital and lunar crew compartments, the baseline approach given in Figure 3-1 can be considered typical of the refurbishment of both types of vehicles.

As shown in Figure 3-1, window panels and panel frames will be removed from the crew compartment for refurbishment in parallel and will be subsequently reinstalled and gasketed. This procedure is in accordance with current Block II practice. Local Class C refurbishment will entail routing out old ablator and gunning and curing of new ablator. The cure cycles to be utilized will be the current 200-210°F cure and a second cure, either the current post-cure or a new cure (to be determined by engineering study) designed to meet ablator shrinkage restraints. Note also that drying of the ablator following machining is assumed to be required.

Figure 3-2 describes the baseline refurbishment of earth orbital recovered aft compartments. Since Class B refurbishment is assumed feasible for earth orbital recovered aft compartments, no Class C refurbishment processes are considered except the preparation of new shear and compression pads. For lunar return aft compartments the process of ablator and pad removal and inspection must be added to the current Block II sequence. All forward compartments, supplied as new hardware for each reuse, will be fabricated with reduced thicknesses in accordance with current Block II sequences.

3.2 BASELINE REFURBISHMENT PROGRAM TASKS

The execution of a development/operations phase program to refurbish the aforementioned five Block II vehicles in accordance with the baseline approach will require the performance of specific engineering, test, and manufacturing tasks.
Table 3-3 gives a preliminary description of the major tasks required to execute baseline refurbishment of these vehicles. These tasks have been presented to conform to the NAA/Apollo task structure:

Task No.	Task Area
1	Integration & Systems Analysis
2	Reliability & Quality Assurance
3	Instrumentation
4	Training and Simulations
5	System Test Operations
6	Launch Operation & Field Support
7	Documentation
8	Program Management
9	Design and Development
10	Preproduction & Rearrangement
11	C/M GSE
12	S/M GSE
13	Fabrication and Assembly
14	Fabrication and Assembly, Spares

Table 3-3 also gives an estimate of the relative level of effort associated with baseline refurbishment and total Class C refurbishment of all five vehicles. In general, reliability, QA, documentation, management, engineering, and test tasks all require significantly greater levels of effort for the baseline refurbishment (due to the use of Class A and B refurbishments) than for an approach restricted to total Class C refurbishment. Fabrication, assembly, and QC efforts are estimated, however, to be greater for the total Class C approach.

3.3 SCHEDULE CONSIDERATIONS

The basic schedule constraint for refurbishment of the command module heat shield is the requirement for delivery of the refurbished heat shield a minimum of five months prior to second launch. This will permit a maximum period of 7 months or 27 weeks for Avco/SSD refurbishment. Figure 3-3 gives a comparison of this constraint with preliminary estimates of baseline refurbishment

FOLDOUT FRAME 1

	Receiving Inspection	Thickness Index & Drill Inspection Holes - Major Dimensional Data	Disassemble, Fit Doors, Hatches, Frames; Remove Window Panels	Machine Ablator ML for reuse	Measure residual Water	Dry	Measure Residual Char, Identify Class C Areas	Rout Charred Ablator	Clean, Prime, Layup, Bond & NDT H/C, Edgemembers	Thickness Inspection
CREW*								C	C	C

PANEL FRAMES -

CLASS A & C REFURBISHMENT

*LETTER INDICATES CLASS OF REFURBISHMENT AREA APPLICABLE; OTHERWISE, ALL AREAS APPLICABLE

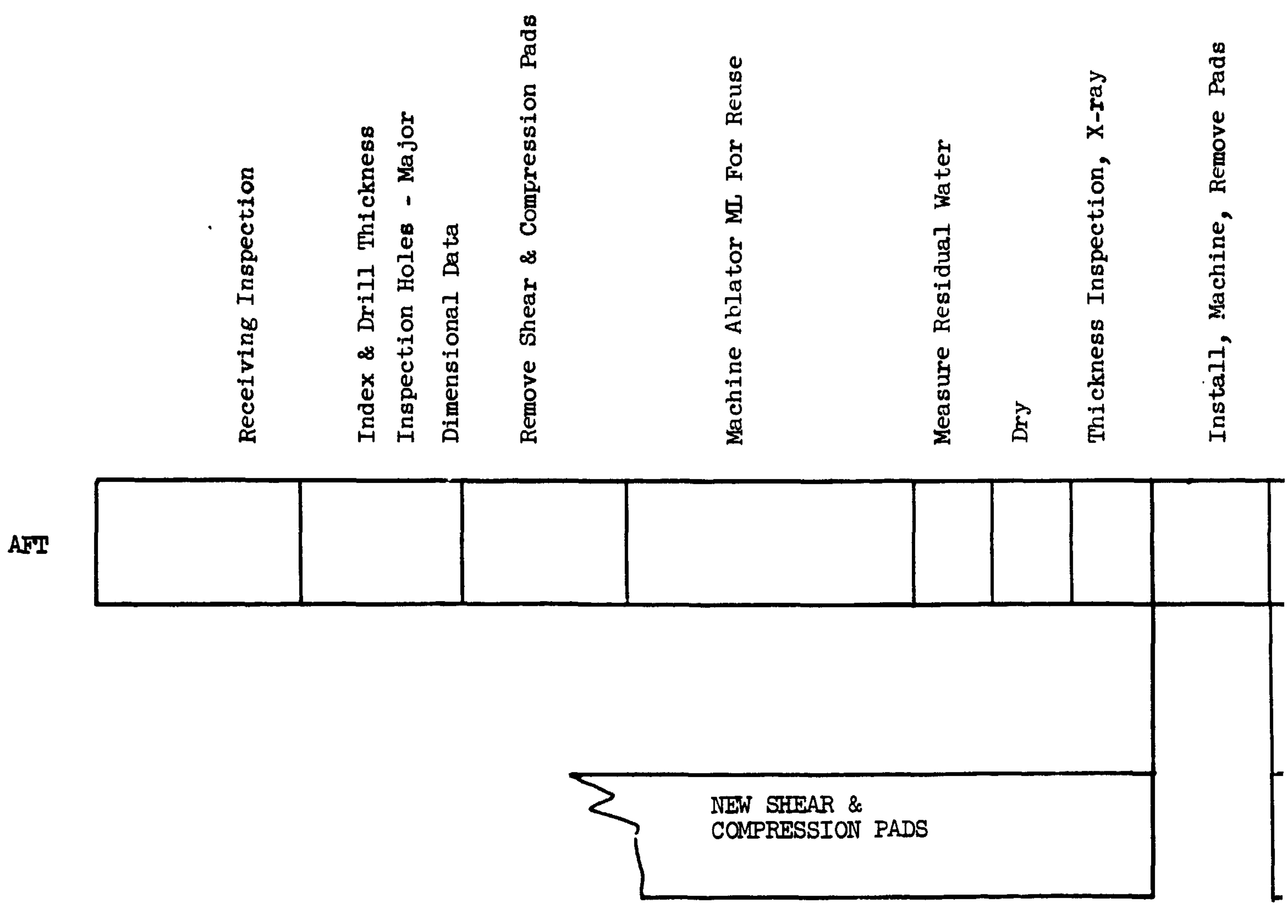
FOLDOUT FRAME 2

Clean, Prime, Layup, Bond & NDT H/C, Edgemembers	Thickness Inspection	Machine H/C, EM	Thickness Insp., Cut Gap Splices	Clean, Prime H/C	Tape, Gun	Vacuum Bag & Cure	Index & Drill Thickness Holes & Insp.	Machine Ablator ML	X-ray, Thickness Inspection, complete all repairs	Vacuum Bag and Cure	Mate Fwd.-Crew; Fair / Interfaces, Repairs; Cast All Gaskets	Final Repairs; X-ray; visual / Dispo 'tion	Pore Seal	Moisture Barrier - Paint	Weight & C. G.	Accept & Ship
C	C	C	C	C	C		C	C	C							

FRAMES - REFURBISHMENT

ISHMENTS - WINDOW PANELS

Figure 3-1 BASELINE REFURBISHMENT - EARTH ORBIT AND LUNAR HEAT SHIELDS - CREW COMPARTMENT


Receiving Inspection
Index & Drill Thickness
Inspection Holes - Major
Dimensional Data
Remove Shear & Compression Pads
Machine Ablator ML For Reuse
Measure Residual Water
Dry
Thickness Inspection, X-ray
Install, Machine, Remove Pads
AFT
NEW SHEAR &
COMPRESSION PADS

FOLDOUT FRAME

FOLDOUT FRAME

FOLDOUT FRAME 2

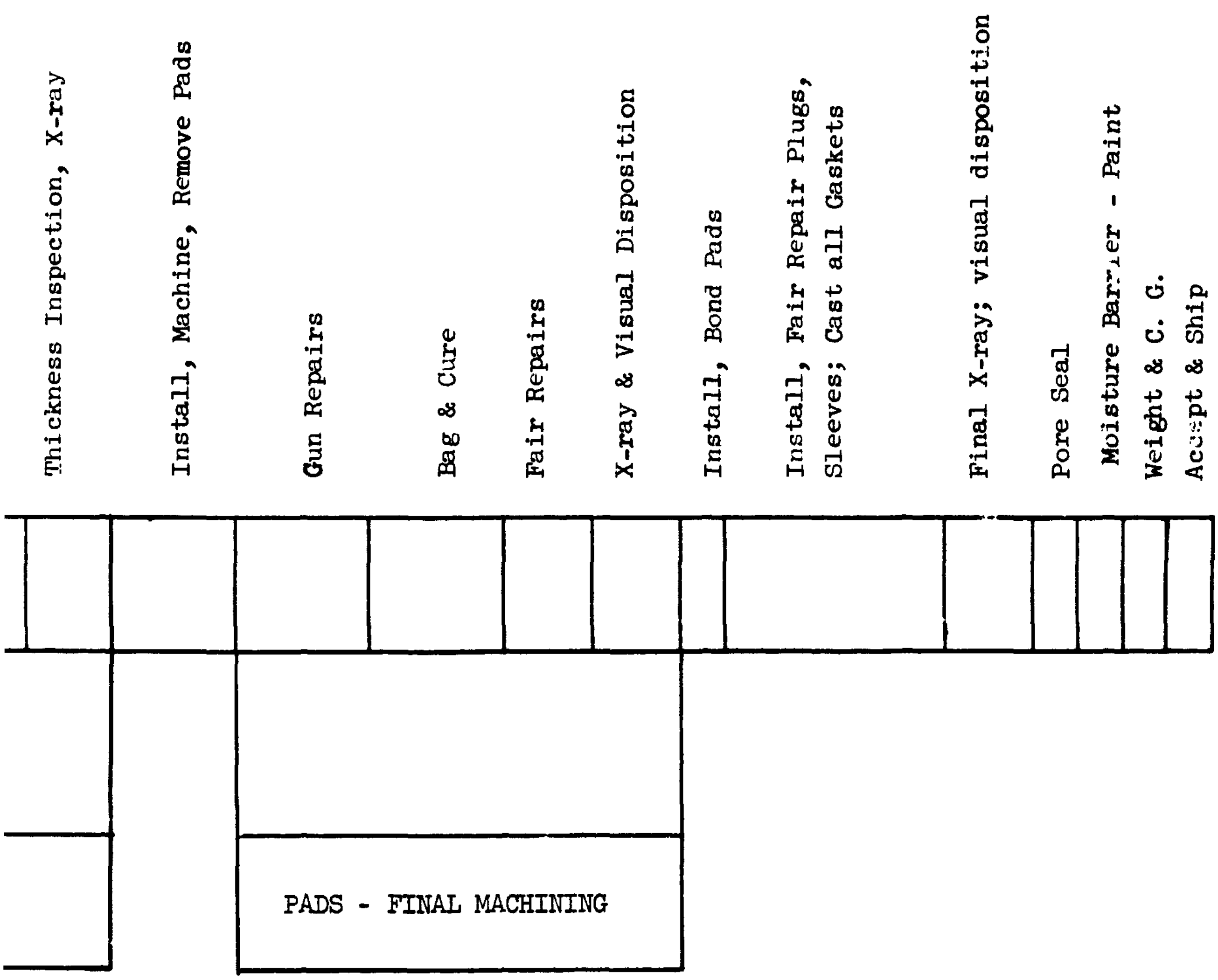


Figure 3-2 BASELINE REFURBISHMENT - EARTH ORBIT HEAT SHIELD - AFT COMPARTMENT

TABLE 3-3

BASELINE REFURBISHMENT PROGRAM TASKS

Task No.	Task Description	Effectivity* Baseline	Effectivity* Total Class C
1.0	Integration and Systems Analysis	NA	NA
2.0	Reliability and Quality Assurance Review and revise reliability and associated plans, procedures, and specifications and conduct quality engineering necessary to assure compliance with reliability and design requirements.	B	A
2.1	Failure Reporting, Corrective Action, Recurrence Control	B	A
2.2	Quality Assurance Provisions Review Quality Assurance provisions of specifications and update by specification change. Review and revise Block II QATP's to meet refurbishment EC requirements, particularly in area of NDT.	B	A
2.3	Quality Engineering Plan detailed test and inspection procedure. Support MRB activity. Perform process control analyses. Prepare end-item acceptance test plan. Conduct vendor surveillance. Perform production scale-up and specification of NDT inspection techniques and processes for immersion water content and residual char: design, fabricate, and/or procure associated tools, fixtures, and equipments.	B	A
2.4	Production Monitoring Plan and perform production monitoring tests to assure material property standards.	B	A
3.0	Instrumentation	NA	NA
4.0	Training and Simulation	NA	NA
5.0	Systems Test Operations (Flight Test Support)	NA	NA
6.0	Launch Operations and Off-Site Support Off-Site support to NAA/S&ID at KSFC and Downey	B	B
7.0	Documentation Issue program documentation in accordance with NAA/S&ID documentation specifications.	B	A
8.0	Program Management Conduct program planning in accordance with NAA/S&ID specifications, including preparation of engineering, test, manufacturing, facilities, configuration management, and support plans as 'delta' documents to Block II plans. Preparation of program progress reports, cost reporting, program review and control.	B	A
9.0	Design and Development Design, develop, and test refurbishment of Apollo Block II heat shield panels for reuse in earth orbit for up to 30 days. Provide performance analyses of refurbished heat shield panels including substructure. Develop and select materials and processes to be used and the test program necessary for obtaining data for use in design and for verification of design performance.	B	A
9.1 9.1.1	Applied Mechanics Thermostructural analyses of design performance of the composite heat shield panels including reused ablator, gasket and other materials, and substructure will be performed. Manufacturing and test support will be provided.	B	A
9.1.2	Structural tests will be performed for obtaining design data and for verification of structural design performance including effects of ablator reuse, water immersion and drying, and 30 day exposure.	B	A
9.2	Aerodynamics	NA	NA
9.3	Materials Develop materials and fabrication processes required for refurbishment of composite ablation heat shield panels and components. Develop NDT and inspection techniques for residual char and immersion water content determinations. Provide manufacturing support.	B	A

*Relative Level of Effort A<B<C

TABLE 3-3 (Concl'd)

Task No.	Task Description	Effectivity*	
		Baseline	Total Class C
9.4	Physics Conduct aerothermodynamic tests for determination of performance and materials data on reused ablator, including effects of prior entry, water immersion, drying, and 30 day exposure. Conduct aerothermodynamic tests for verification of performance of local areas subject to configuration change, e.g., protruding shear and compression pads.	B	A
9.5	Design Provide design changes to Block II configuration to account for ablator thickness changes, altered heating and perturbations, substructure design changes, NAA/Avco interface changes, and changes to provide for refurbishment and reuse. Provide manufacturing and weight-c.g. support.	B	A
9.6	Thermodynamics Perform aerothermodynamic analyses of design performance of composite heat shield panels revised ablator, gasket and other materials, and substructure. Generate ablator design thicknesses for reuse mission. Provide manufacturing and test support	B	A
10.0	Preproduction and Rearrangement	NA	NA
11.0	C/M GSE	NA	NA
12.0	S/M GSE	NA	NA
13.0	Fabrication and Assembly Refurbish, assemble, acceptance test, and deliver five (5) heat shield ablative panel subsystems. Design, fabricate and/or procure those items of special tooling required for fabrication and assembly.	B	C
13.1	Manufacturing	B	C
13.1.1	Manufacturing Engineering Conduct Manufacturing program coordination, QC and engineering liason, production and material control, and numerical control tape generation.	B	A
13.1.2	Tooling Maintain and update Block II tooling. Design, fabricate, and/or procure special manufacturing tooling required for refurbishment.	B	A
13.1.3	Raw Materials Procure and process raw materials for shield fabrication and assembly.	B	C
13.1.4	Ablator Material Preparation Prepare 5026-39H/C-G and 5026-39M materials for use in ablator refurbishments. Prepare pore seal and moisture barrier-paint materials for refurbishment and finishing.	B	C
13.1.5	Detail Parts Fabrication Fabricate detail heat shield parts and components including edgemembers shear and compression pads, abort tower wells, ablator bolt plugs, repair plugs, and several catagories of miscellaneous parts.	B	C
13.1.6	Refurbishment and Assembly Perform refurbishment and assembly operations from receiving and receiving inspection, through ablator removal and drying, through refurbishment and assembly and, finally, through shipment.	B	C
13.2	Quality Control	B	C
13.2.1	QC Engineering Conduct QC program coordination, manufacturing, and engineering liason, and MRB.	B	A
13.2.3	Receiving Inspection tests Conduct raw materials, parts, and component receiving inspection tests.	B	C
13.2.4	In-Process Inspection and Surveillance Tests Conduct in-process inspection and surveillance tests of ablator preparation, detail part fabrication, and heat shield refurbishment and assembly operations.	B	C
13.2.5	Acceptance Tests Conduct acceptance of components and detail parts.	B	C
13.2.6	End-Item Acceptance Tests	B	B
14.0	Fabrication and Assembly, Spares	NA	NA

*Relative Level of Effort: A<B<C

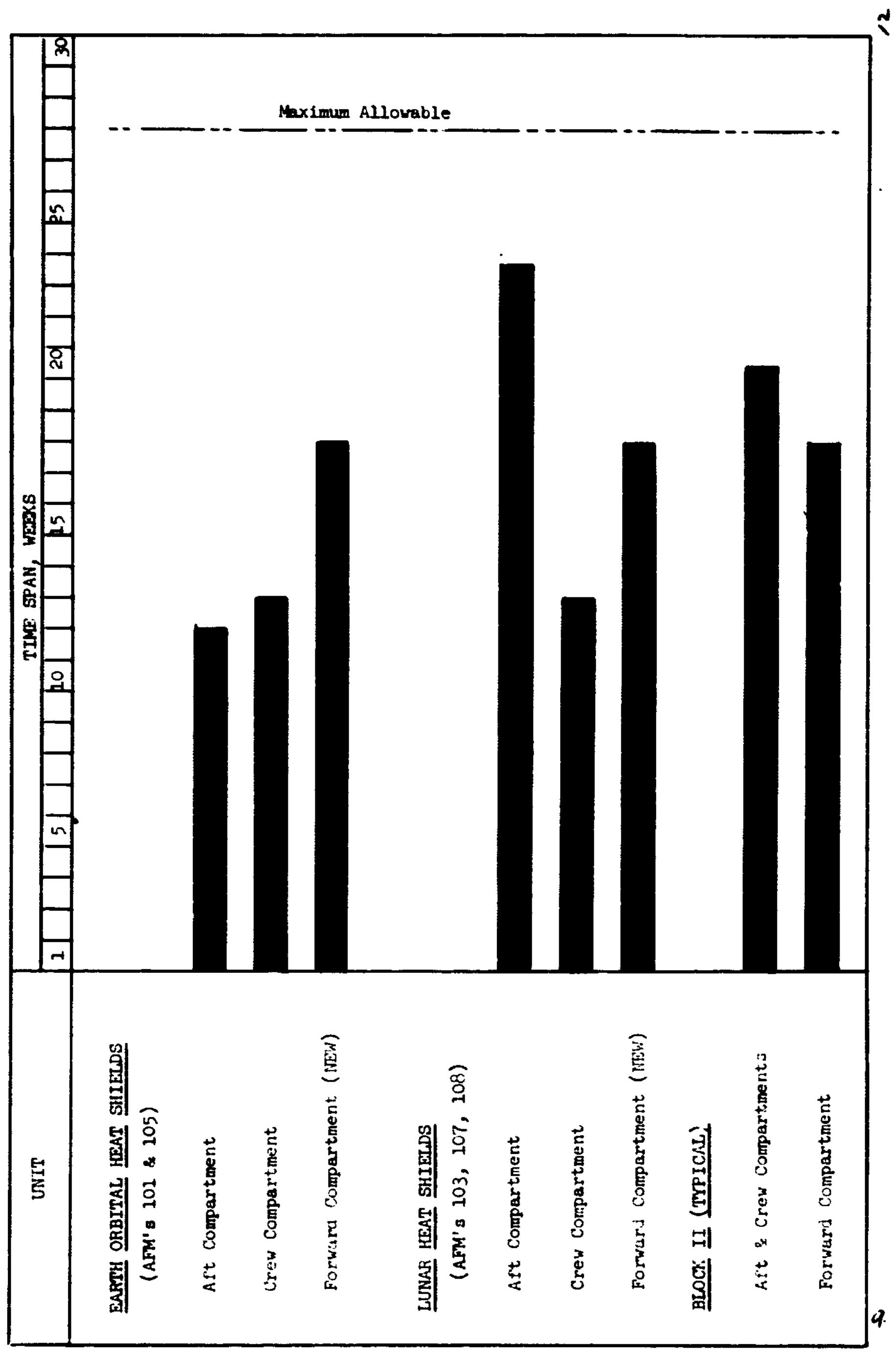


Figure 3-3 PRELIMINARY BASELINE REFURBISHMENT TIME SPANS

time spans. As shown in Figure 3-3, preliminary estimates indicate that baseline refurbishment should be possible to accomplish within the 27 week constraint. The longest time span appears to be associated with a total Class C refurbishment of the aft compartment. (A similar time span would be required for a total class C refurbishment of the crew compartment.)

It is very important to caution that the time spans presented in Figure 3-3 are preliminary estimates, typical only of the assumed baseline refurbishments. These estimates are subject to the validity of underlying assumptions specified in Section 3.0 and subject to individual vehicle variations. The validity of Class A and B refurbishments has not been demonstrated.

A further constraint on the baseline refurbishments is the requirement to mate the crew and forward compartments for fairing interfaces and for casting the intercompartmental gasket. For baseline refurbishment, the maximum advantage under this constraint would be obtained by early receipt of the forward compartment substructure. Such an approach could significantly advance the delivery of the earth orbital crew compartment which would otherwise require a hold. Subsystem delivery of refurbished lunar heat shields is constrained by the aft compartment. Baseline Class C refurbishment of the aft compartment is expected to require 3-4 weeks greater time than a new Block II because of the requirement to inspect and remove used ablator prior to a standard Block II fabrication. Should the lunar crew compartment also require a total Class C refurbishment, the time span for its refurbishment would be essentially the same as that for the lunar aft compartment.

4.0 COST CONSIDERATIONS

Almost all the factors presented in this report as affecting the feasibility or execution of refurbishment also directly affect the cost of refurbishment. In particular, the assumptions that have been required to generate the baseline plan described in Section 3.0 are so numerous and so fundamental that the validity of any cost estimated for the baseline plan is wholly dependent upon the validity of these assumptions. In addition, other factors such as the actual state of individual recovered vehicles, the amount of actual engineering, development, and test required to demonstrate and verify reuse designs, and the potential of undiscovered problems will have significant effects on baseline unit cost. Because of the tenuousness of these assumptions and because of the lack of adequate definition of these other factors, no firm unit cost data are available at this time.

In an attempt to obtain some feeling for cost considerations associated with refurbishment, preliminary, crude cost comparisons were generated in this study. The baseline refurbishment program outlined in Table 3-3 was used as the basis for this comparison. Table 3-3 presents a qualitative comparison at the levels of effort required to execute two refurbishment programs.

As a result of a preliminary, budgetary analysis of these two programs, the crude estimates in Table 4-1 were obtained for the relative costs of these two programs.

TABLE 4-1

PRELIMINARY COMPARISON OF RELATIVE COSTS OF REFURBISHMENT APPROACHES

Task	Baseline Cost (Percent of Total Class C)
2.0 Reliability & Quality Assurance	120
6.0 Launch Operations & Off-Site Support	100
7.0 Documentation	130
8.0 Program Management	110
9.0 Design and Development	170
13.0 Fabrication and Assembly	80
Total Program	190

It is important to point out that the uncertainty in the estimates in Table 4-1 is at least ±25%. On the basis of these preliminary estimates it appears that baseline refurbishment of the five vehicles considered in this study might yield a 10% saving over total class C refurbishment. For a larger number of vehicles the saving might approach 20%. In view of the uncertainty in these estimates, the difference of 10% between baseline and total class C refurbishment is insignificant. Until a better definition of fundamental cost factors becomes available it can only be concluded that the cost savings of baseline refurbishment may be insignificant.

5.0 SUMMARY AND CONCLUSIONS

In Summary, this preliminary study has indicated the following:

1) Class C refurbishment, complete replacement of used ablator with new ablator, appears technically feasible

2) The feasibility of Class A and Class B refurbishments has not been demonstrated

3) Demonstration of the feasibility of Class A and Class B refurbishments requires the following:

   a) Detailed analyses of post-flight data on C/M 009, 011, 017, and 020 to obtain a realistic appraisal of char depth distribution and of local anomalies

   b) Demonstration of a reliable production technique for determining char depth or the existence of residual char on machined ablator surfaces

   c) Definition of a performance-acceptable degree of degradation in reused ablator

   d) Assessment of the effect of prior entry exposure on the thermo-structural and aerothermodynamic performance of used ablator

   e) Demonstration of a reliable production technique for measuring the amount of salt water retained in ablator materials destined for reuse

   f) Assessment of the effects of residual salt water, retained salt, and drying processes on ablator thermostructural and aerothermodynamic performance

   g) Demonstration of the ability to detect bond delamination in reused ablator which has undergone water impact or other structural or thermostructural loading

4) No design factors have been identified which constrain the feasibility of Class A, B, or C refurbishments

5) No manufacturing process has been identified in Class A, B, or C refurbishment which requires manufacturing skill or equipment not already developed and on use at Avco/SSD on the Block II Apollo program

6) The acquisition of the following post-flight test data from AFM 017 and 020 is expected to significantly enhance the assessment at feasibility of Class A and B refurbishments:

   a) Density-temperature gradients

   b) Water content measurements

   c) Char depth distribution

7) The concept of identification of char depth and selection of final local refurbishment class after machining ablator to reuse thickness is proposed. This approach is considered less costly and time consuming than that of determining char depth and approach prior to machining

8) Baseline refurbishment should be possible in significantly less time than the 27-week constraint. Earth orbital baseline refurbishment with early forward compartment delivery may be possible in less than 15 weeks. Lunar baseline refurbishment may require greater time than current Biock II because of the requirement for total Class C refurbishment of the aft compartment

9) Preliminary cost estimates indicate that baseline refurbishment of the five subject vehicles may yield a 10% saving over total Class C refurbishment. In view of the uncertainty in these estimates, it can only be concluded at this time, without further study, that the cost saving is insignificant